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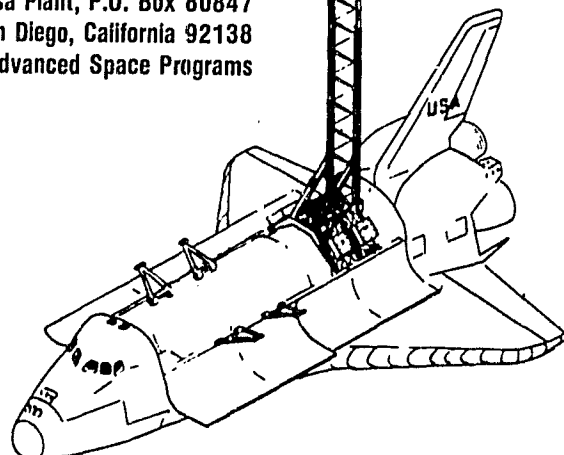# SPACE CONSTRUCTION EXPERIMENT DEFINITION STUDY (SCEDS) PART II

FINAL REPORT

VOLUME II • STUDY RESULTS

CONTRACT NO. NAS9-16303
DRL NO. T-1346
DRD NO. MA-664T
LINE ITEM NO. 3

GENERAL DYNAMICS
Convair Division

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Advanced Space Programs

SPACE CONSTRUCTION EXPERIMENT DEFINITION STUDY (SCEDS) PART II

FINAL REPORT

VOLUME II • STUDY RESULTS

26 April 1982

FOREWORD

The final report was prepared by General Dynamics Convair Division
for NASA/JSC in accordance with Contract NAS9-1603, DRL No.
T-346, DRD No. MA-664T, Line Item No. 3. It consists of two vol-
umes: (I) a brief Executive Summary and (II) a comprehensive set
of Study Results.

The principal study results for Part II of the Space Construction
Experiment Definition Study (SCEDS) were developed from September
1981 through February 1982, followed by final documentation. Re-
views were presented at JCS on 17 December 1981 and 2 March 1982,
and at NASA Headquarters on 4 March 1982.

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LIST OF ACRONYMS AND ABBREVIATIONS

A/D	Analog-to-Digital
AFD	Aft Flight Deck
ASE	Airborne Support Equipment
C&W	Caution and Warning
CCD	Charged Coupled Device
CCLS	Computer Controlled Launch Set
CCTV	Closed-Circuit Television
CDR	Critical Design Review
CER	Cost Estimating Relationship
CITE	Cargo Integration Test Equipment
CRT	Cathode Ray Tube
CSDL	Charles Stark Draper Laboratory
CU	Control Unit
CWES	Caution Warning Electric Assembly
DAP	Digital Automatic Pilot
DEU	Display Electronics Unit
DIO	Discrete Input-Output
DRD	Data Requirements Document
DRL	Data Requirements List
EVA	Extravehicular Activity
F/D	Focal Length/Diameter
FSE	Flight Support Equipment
FSS	Flight Support System
F/T	Failure Tolerant
GPC	General Purpose Computer
GSE	Ground Support Equipment
IECM	Internal Environment Contamination Monitor
JSC	Johnson Space Center
KSC	Kennedy Space Center
LaRC	Langley Research Center
LSS	Large Space System

LIST OF ACRONYMS AND ABBREVIATIONS, Contd

MDF	Manipulator Development Facility
MDM	Multiplexer/De-Multiplexer
MMSE	Multiuse Mission Support Equipment
MU	Multiplexer
NASA	National Aeronautics and Space Administration
O&C	Operations and Checkout Facility
OPF	Orbiter Processing Facility
OTV	Orbital Transfer Vehicle
PCM	Pulse Code Multiplexer
PDI	Payload Data Interleaver
PDR	Preliminary Design Review
PMP	Parts, Materials, and Processes
PPF	Payload Processing Facility
PRCS	Primary Reaction Control System
PROM	Programmable Read Only Memory
PRR	Preliminary Requirements Review
RAM	Random Access Memory
RCS	Reaction Control System
RMS	Remote Manipulator System
RSS	Rotating Service Structure
SCE	Space Construction Experiment
SCEDS	Space Construction Experiment Definition Study
SIO	Serial Input-Output
SSP	Standard Switch Panel
STP	System Test Plan
STS	Space Transportation System
VAB	Vertical Assembly Building
VPF	Vertical Processing Facility
VRCS	Vernier Reaction Control System
WBS	Work Breakdown Structure
WETF	Weightless Environment Test Facility

SECTION 1

INTRODUCTION

1.1 SCOPE

This is the second of two volumes comprising the SCEDS Final Report. It contains the detailed results of all Part II study tasks. Volume I provides an Executive Summary of the study results. This report is the final deliverable contract data item. It satisfies the requirement for Line Item 3 (DRD MA-664T) of DRL T-1346.

1.2 STUDY OVERVIEW

1.2.1 PART I SUMMARY. The Part I study tasks focused on the definition of a baseline Space Construction Experiment (SCE) concept, shown in Figure 1-1 and concepts for additional suitcase experiments for Extravehicular Activity (EVA) and Remote Multipulator System (RMS) construction operations to incorporate the following characteristics:

a. Test a representative Large Space System (LSS) element. The baseline experiment employs a 50m deployable low natural frequency structure. The structure has a very low coefficient of thermal expansion, achievable through the use of graphite composite materials for construction. Structural dynamic tests provide data to be correlated with math model predictions. Minimal ground testing is to be performed, and minimum flight instrumentation employed.

b. Share a Shuttle mission with other payloads as a payload of opportunity.

c. Remain attached to the Orbiter throughout the test. Jettison capability is provided; however, the experiment will normally be automatically retracted, restowed, and returned to earth by the Orbiter.

d. Provide options to approach proven capabilities of the Orbiter conservatively and safely exceed proven limits to establish usable capabilities for control, mission timelines, and critical interfaces. These options include variability of mass moment of inertia and variable damping augmentation.

e. Exercise a variety of appropriate Large Space System (LSS) construction and assembly operations utilizing basic Space Transportation System (STS) capabilities (EVA, RMS, CCTV, Illumination, etc.) to be correlated with ground tests and simulations.
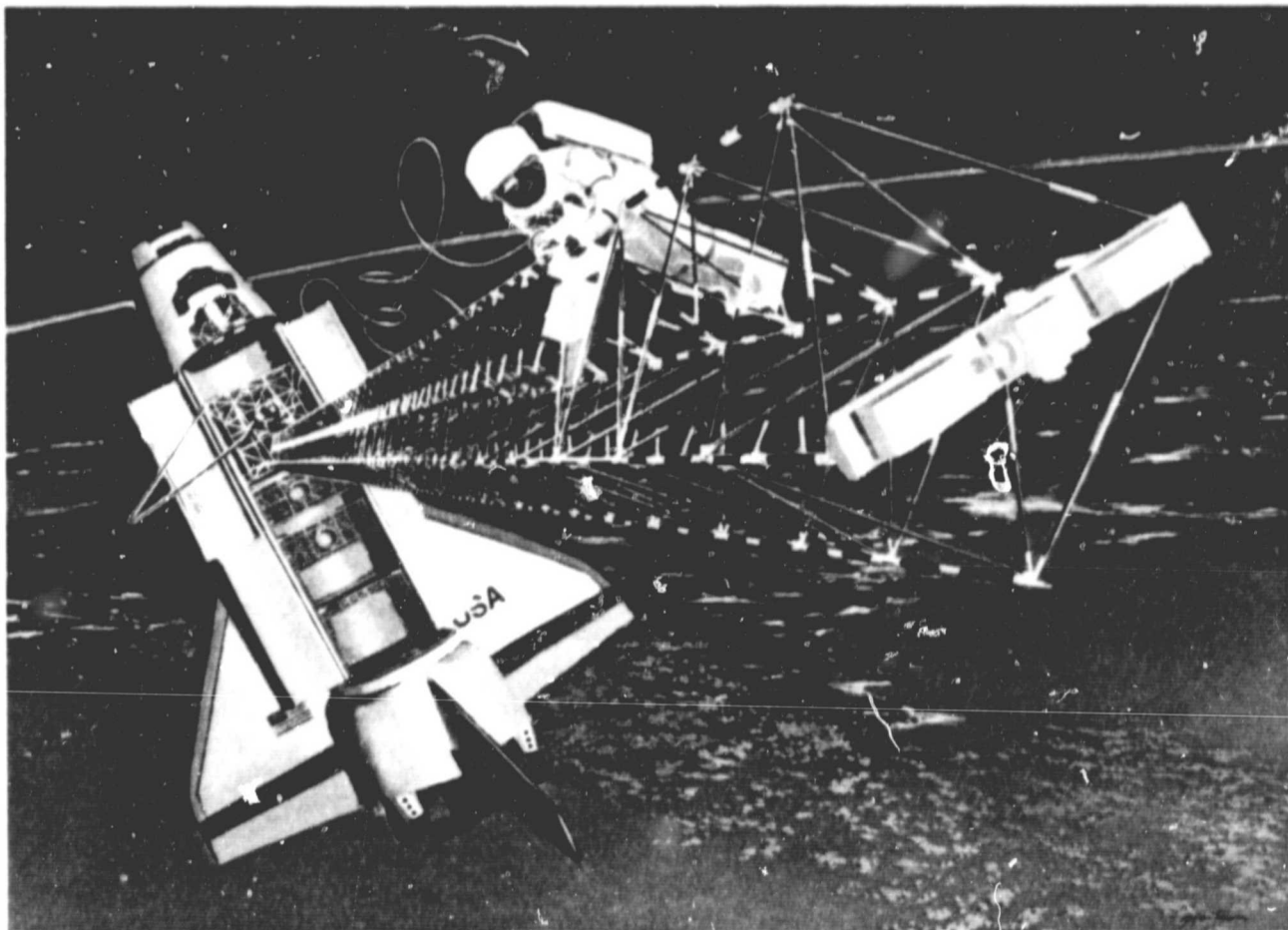
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Figure 1-1. Baseline Flight Experiment Concept

The principal Part I conclusions applicable to Part II are summarized in Table 1-1.

1.2.2 <u>PART II SUMMARY</u>. After the conclusion of Part I, the study objectives were expanded by NASA JSC and LaRC to place greater emphasis on the structural dynamics and controls technology aspects of the experiments and to specifically design the experiment to develop and demonstrate the technologies to meet requirements for large space antenna feed masts. The objectives continued to stress the development of Orbiter capabilities necessary to support large space structures construction operations, including the ability to maneuver and control large attached structures and to perform in-space deployment and construction operations.

The Part II study activities were divided into five major tasks which interrelated as shown in Figure 1-2. Task I further developed and defined the SCE for integration into the Space Shuttle. This included development of flight assignment data, revision and update of preliminary mission timelines and test plans, analysis of flight safety issues, and definition of ground operations scenarios.

Table 1-1. Part I Principal Conclusions

STRUCTURE

- Tetrahedral diamond cross-section truss beam best overall LSS applicability

- Safety dictates controlled linear deployment and retraction

- Near-zero-coefficient of thermal expansion (CTE) achievable with graphite/epoxy beam construction

- PRCS contingency loads penalize beam flexibility, cost, weight and packaging efficiency

- Semi-precision test structure less costly and allows assessment of rattle and backlash effects

DYNAMICS & CONTROL

- DAP not challenged by baseline configuration

- Flexible base mount needed to produce low modal frequencies that challenge the DAP

- Torque wheel actuators at beam tip provide variable beam damping and excitation capability

ORBITER/MISSION INTEGRATION

- Flight assignment needed to confirm configuration

- Two day experiment optimum

- EVA experiments time limited

- RMS has a handling problem during jettison due to beam tip mass

PROGRAMMATICS

- 1984-1985 flight achievable

- Total payload cost: $10M

Task 2 incorporated new requirements for the flight experiment and defined changes to satisfy these requirements for a large space antenna feed mast test article as well as more detailed structural dynamics and controls experiments.
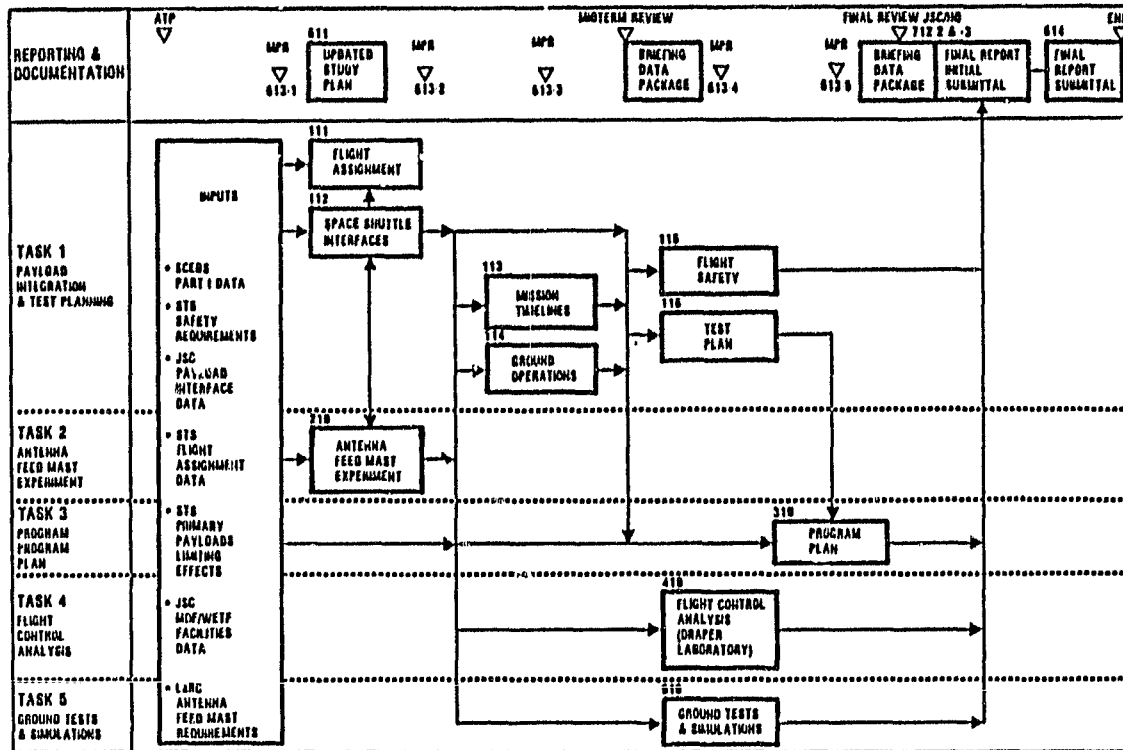
Figure 1-2. SCEDS Part II Task Relationships

Task 3 expanded and updated the Part I preliminary program plan and cost estimates based on the revised preliminary design data and test plan.

In Task 4, Convair provided revised SCE structural dynamic characteristics to the Charles Stark Draper Laboratory for simulation and analysis of experimental tests to define and verify control limits and interactions effects between the SCE and the Orbiter Digital Automatic Pilot (DAP).

SECTION 2

FLIGHT EXPERIMENT TEST ARTICLE

In SCEDS Part I a basic experiment was defined to satisfy general guidelines established by the NASA/JSC. The baseline deployable structure test article was selected for its applicability to numerous large space systems, including platforms and antennas. The NASA Langley Research Center (LaRC) recognized the potential benefits of the basic flight experiment as an evolutionary step in developing major areas of technology for large space antenna systems. A joint effort by JSC and LaRC was initiated to expand the objectives and detail requirements for the SCEDS Part II. This section presents the new SCE system requirements and describes the changes made to the structural test article to meet these requirements.

2.1 OBJECTIVES AND REQUIREMENTS

Five major areas of technology investigation will be explored by the SCE program:

a. Prediction of, and inflight identification of, structural dynamic parameters for large deployable structures.

b. Validation of inflight modal damping controls for large space structures.

c. Development and verification of technologies to meet large space precision antenna/feed mast requirements.

d. Confirmation of Orbiter flight control system modeling and simulation techniques for predicting DAP performance with a large flexible structure attached to the Orbiter.

e. Utilization of Orbiter capabilities to perform large space system construction and assembly operations.

The following objectives and requirements submitted as inputs to Part II of SCEDS expanded the area of application of the basic beam defined in the Part I study.

2.1.1 EXPERIMENT OBJECTIVES - STRUCTURAL DYNAMICS & CONTROL. The detailed experiment objectives in the areas of structural dynamics and structural control are as follows:

a. To determine the degree to which theory and ground testing can predict flight performance with and without damping augmentation.

b. To evaluate math modelling of large, complex, lightweight structures for which ground test results are questionable or unattainable.

c. To evaluate system identification and state estimation algorithms on complex lightweight structures in real space environment. Compare on-line with off-line methods and evaluate errors due to sensor number reduction.

d. To evaluate joint and cable damping effects in zero-g relative to one-g.

e. To evaluate control of structure interactions including concentrated local controller effects on dimensional stability.

f. To evaluate sensor/measurement techniques applicable to low frequency systems with low motion/deflection tolerances.

g. To evaluate deployment dynamics in zero-g vs. one-g.

h. To improve predictability of on-orbit loads on lightweight complex structures.

i. To simulate problems and phenomena expected in very large antennas of 100m class.

j. To establish correlation between ground and spaceflight tests.

2.1.2 EXPERIMENT REQUIREMENTS - STRUCTURAL DYNAMICS & CONTROL.
The flight experiment test article requirements to support structural dynamics and controls investigations are as follows:

a. The test article shall be a many-member/many-joint deployable structure.

b. The test article shall be a low-frequency structure of minimum weight design with a 0.05 to 0.10 Hz cantilever first mode natural frequency.

c. The test article shall have at least two closely coupled modes and preferably more.

d. An instrumentation and data acquisition system capable of measuring deployment motions shall be incorporated in the flight experiment.

e. The test article shall provide excitation for the first four modes, and instrumentation and data acquisition for recording mode shape and frequency response of the first six modes and attachment loads of Orbiter.

f. A means for studying the effect of joint loading shall be provided.

g. A control device (or devices) for demonstration of modal damping and critical gain determination shall be provided.

h. Nonlinearities shall be kept to minimum except where included as deliberate superposition for test purposes.

i. Force and motion measurement at Shuttle interface shall be provided.

j. Tension loading (or other methodology) to vary properties for part of or the entire mast shall be provided.

k. Measurement of Shuttle Orbiter motions shall be provided.

All but two of the above requirements were incorporated into the SCE. Requirement c. was investigated (see subsection 4.2.6). The capability to produce two or more closely coupled modes was excluded from the preliminary design because of its added cost and because these effects can be investigated through ground tests. Requirement j. was not incorporated because techniques for varying structural properties were not defined.

2.1.3 **FEED MAST STRUCTURAL REQUIREMENTS**. The structural requirements for the feed mast test article as established by NASA/LaRC are shown below. The goal was to achieve a mast structure which fell within size and stiffness parameters considered appropriate for large space antenna feed mast structures.

a. Size: Length = 100m
        Depth = 1.8 to 2.8m

b. Stiffness: Approximately 2×10^7 N-m^2

c. Tip position criteria: ± 10cm longitudinal deviation
                          ± 10cm combined rotation/translation

d. Linear compaction ratio: $\dfrac{\text{deployed length}}{\text{stowed length}} = 20 \text{ to } 25$

e. Test article design to withstand vernier RSC in lieu of primary RCS worst case dynamic loading conditions

2.2 MAST STRUCTURE

The baseline structural test article selected in Part I of the study is the Convair designed deployable tetrahedral truss with a diamond cross section. The baseline configuration is 50m long, 1.61m × 2.28m in depth, with a stiffness of 6.1 × 10^7 N-m^2 in pitch and 2.936×10^7 N-m^2 in roll. It has a 8.9:1 linear compaction ratio and employs a 400 kg tip mass. It's minimum first mode bending frequency for the rigid Orbiter-attached free-free condition is 0.20 Hz.

Potential feed mast candidates were compared before defining the feed mast test article. The selected mast structure was then sized and analyzed for dynamic performance.

2.2.1 **FEED MAST CANDIDATES.** The structural candidate trades (reference 1) of the SCEDS Part I compared ten concepts. Hinged longeron concepts are the most viable feed mast candidates. The three distinct types of hinged longeron structure are represented by the diamond beam, the half-diamond beam, and the delta beam, all of which are shown in Figure 2-', These structures are compared in Table 2-1.

The diamond truss has a number of advantages which make it most suitable for an experimental program.

a. Its low volume packaging requires less cargo space in the Shuttle Orbiter payload bay which allows it to "piggyback" other palletized payloads.

b. Its single failure tolerant structure is an important safety consideration because of the potential for damaging thin walled slender struts during EVA and RMS activities.

c. The all-rigid-strut construction provides greater confidence that the structural properties will remain as modelled throughout ground and flight testing.

By changing the geometry, a diamond truss can be produced with a square cross section. A square truss would provide approximately the same bending stiffness in both pitch and roll, but offers no particular experimental advantage and requires more material to produce.

The half-diamond truss looks attractive from the standpoint of low packaging volume and lower cost than the diamond truss. However, to achieve the best linear packaging ratio, the open square sections on one side would have to be stabilized with diagonal cable members. This non-uniform cross-section may cause torsional shape distortions in bending and provide a more complex model to deal with experimentally.

The delta truss has the best linear packaging density potential. One bay can be folded to a length of two times the diameter of the longitudinal member. General Dynamics has developed concepts for this truss during previous studies and has shown it to be highly advantageous for long antenna masts where stowage length is limited. The necessity for diagonal cables and the large lateral packaging envelope are major disadvantages. The potential problems with diagonal cables are as follows:

a. They have a tendancy to loop and snag. Although careful prepackaging of cables can prevent problems during deployment, the retraction cycle would be subject to cable hang-ups.

b. Cables reduce bending and torsional stiffnesses and strut sizes must be increased to compensate for their tension loads.

Tetrahedral Truss

- **Baseline diamond truss**
- **Square truss**

Delta truss

Half Diamond Truss

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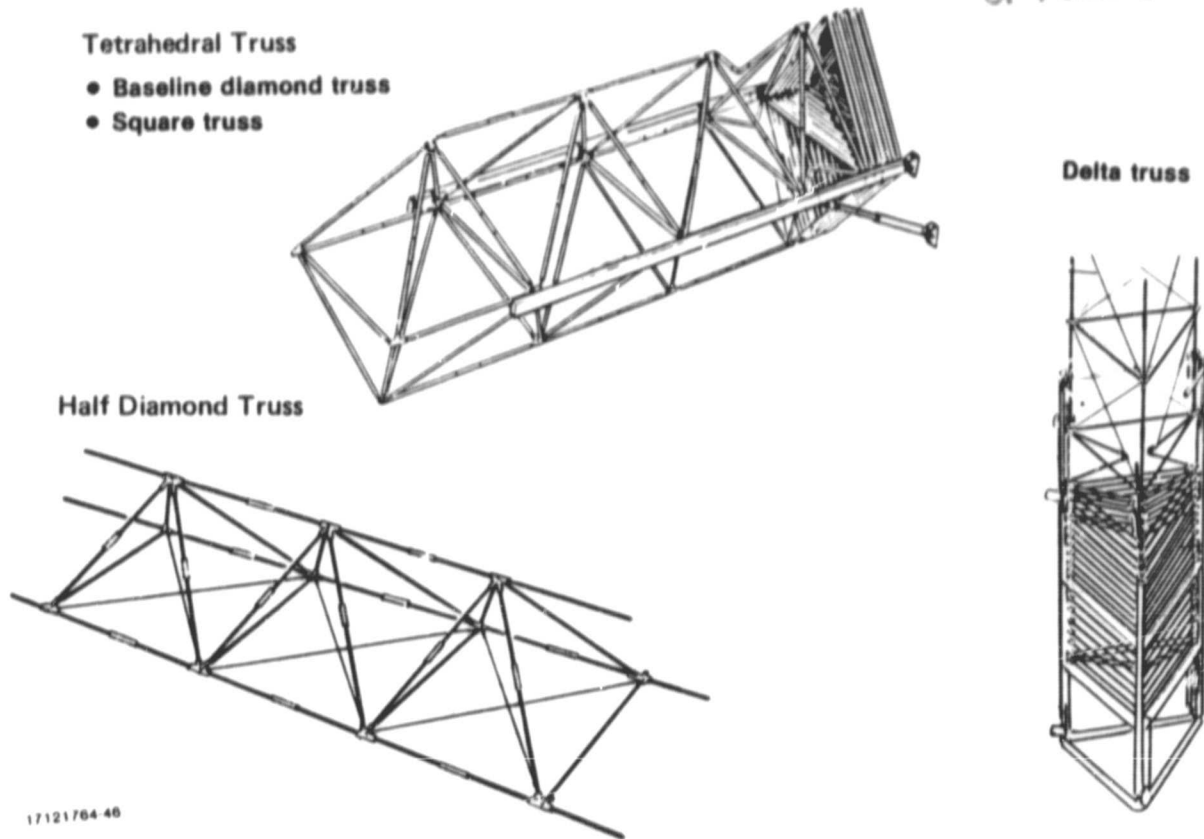
Figure 2-1. Feed Mast Structural Candidates

c. Cables are less durable and more prone to damage from human and mechanical contact. Unreported collisions with cables can result in reduced strength and tension. This may lead to untimely cable failures and unaccountable variances in the structural model.

d. Cable tension control and verification are a major complexity in assuring straightness in a long truss structure.

e. There are several technology problems associated with composite material cables:

   1. Woven or braided composite cables have a tendency to creep and relax their tension loads.

   2. Bundled fiber composite cables fracture easily in bending.

   3. End terminations tend to fracture cable fibers locally, creating weak spots. The integrity and quality of end termination joints require special NDT techniques.

   4. Packaging of cables in the folded structure must be carefully controlled to prevent cable damage from tight bend radii or kinking.

Table 2-1. Comparison of Feed Mast Structural Candidates

Candidate	Advantages	Disadvantages
Diamond Truss	• Packages Flat (Low Volume) • Redundant Structure • Fixed Structural Proper-ties • Two Rail Deployment Mechanism	• 4d Longitudinal Pack-aging • Higher Cost & Weight (More Struts & Fittings)
Square Truss	Same as Diamond Truss	Same as Diamond Truss Except • Requires More Material to Produce
Half Diamond Truss	• Packages Flat (Low Volume) • Lower Cost & Weight (Fewer Struts & Fittings) • Two Rail Deployment System	• 4d Longitudinal Pack-aging • Diagonal Cables One Side • Non-Uniform Cross-Section Properties • Non-Redundant Structure
Delta Truss	• 2d Longitudinal Pack-aging • Lower Cost & Weight (Fewer Struts & Fittings)	• Large Packaged Volume • Diagonal Cables All Sides • Non-Redundant Structure • Additional Deployment Rail & Rail Support Structure

2.2.2 FEED MAST SIZING ANALYSIS. A decision to continue using the diamond truss for the experiment structure was supported by an analysis to verify that it could meet the structural and dynamic requirements for a feed mast. This is an iterative process as illustrated in Figure 2-2. The steps used to size the mast structure are as follows:

a. Select truss characteristics - The factors that directly affect mast sizing are: the diameter, wall thickness, length, and material of the struts; the cross section dimensions; and the tip mass. These parameters define the
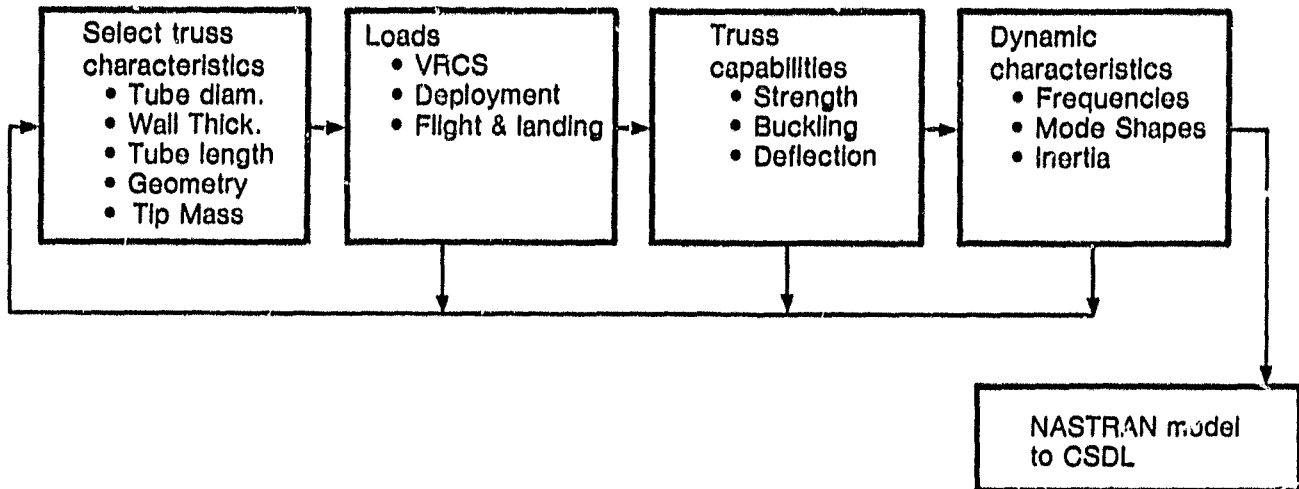
Figure 2-2. Mast Structural Sizing Analysis Flow

strength, stiffness, and mass properties of the truss. Strut diameter and truss geometry are selected to meet the linear packaging ratio required (20:1 ratio was selected).

b. Composite strut loads - Truss loads are estimated for several flight conditions. For on-orbit test conditions, the maximum dynamic moment and shear loads on the structure are calculated in pitch and roll for intermediate and fully deployed states, using a NASTRAN model of the structure and its support system.

c. Analyze truss capabilities - Truss loads are used to compute the allowable stresses and buckling strength capabilities of the truss struts.

d. Composite dynamic characteristics - The MSC/NASTRAN model of the structure is used to compute modal frequencies of at least the first six modes.

The above steps were iterated until an acceptable set of structural characteristics was defined. The selected configuration was then input to a magnetic data tape and transmitted to the Charles Stark Draper Laboratory for evaluating Orbiter Digital Automatic Pilot (DAP) interactions and refined loads analysis. The results of the CSDL analysis are not yet complete.

2.2.3 FEED MAST STRUCTURAL AND DYNAMIC CHARACTERISTICS. The revised geometry of the diamond truss is shown in Figure 2-3. The effects of the preliminary sizing analysis using a 250 kg tip mass are as follows:

a. Width increased from 1.61m to 2.0m

b. Height increased from 2.28m to 2.83m
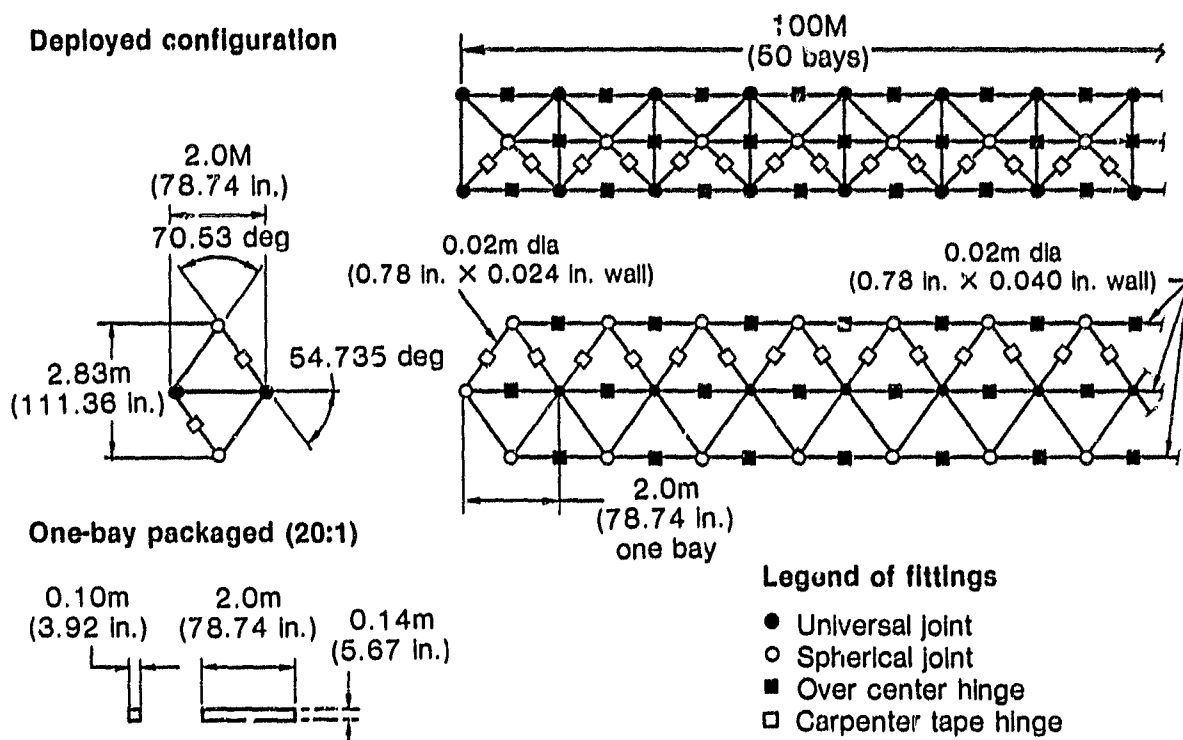
Figure 2-3. Revised Tetrahedral Diamond Truss Geometry

c. Linear packaging ratio increased from 8.7:1 to 20:1

d. Diameter of longitudinal members decreased from 0.045m to 0.02m

e. Length increased from 50.1m to 100m

f. Number of bays increased from 31 to 50

g. Stiffness (EI) in pitch decreased from 6.1×10^7 N-m^2 to 2.0×10^7 N-m^2

h. Stiffness (EI) in roll decreased from 2.9×10^7 to 1.3×10^7 N-m^2

The frequencies for the first six modes are listed in Table 2-2 for two different tip masses with both flexible and rigid support. These four configurations were submitted to CSDL to obtain comparative results on their effects on the DAP (see subsection 4.1).

Table 2-2. Diamond Truss Structural Dynamic Characteristics
(Orbiter-attached Free-Free)

Mode	Description	Frequencies (Hz)			
1	1st pitch bend	.0390	.0861	.0391	.1192
2	1st roll bend	.0618	.1138	.0533	.1322
3	2nd roll bend	.6716	.9350	.6677	.9584
4	2nd pitch bend	.8069	1.1783	.8116	1.2092
5	3rd roll bend	2.2826	2.8937	2.2678	2.9298
6	1st torsion	2.7943	3.1956	2.7901	3.1956

Tip mass (kg)	250	250	100	100
Support stiffness (n/m)	1.55×10^5	∞	$.75 \times 10^5$	∞

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At the end of Part I it was concluded that a flexible base mounting for the flight experiment truss was the best approach to reducing the first mode bending frequency to less than 0.05 Hz to provide a challenge to the DAP. Flexible mounting of the truss may be accomplished as shown in Figure 2-4 by mounting the deployment rail and its supports on a pivoting frame suspended on flat springs for roll flexibility. Helical springs installed in the pitch braces will provide flexibility in pitch. This concept was further developed to install latches and actuators which will allow the mounting flexibility to be locked out so that truss structural dynamic characteristics can be tested. The flex mount would then be unlocked to perform DAP interaction testing.
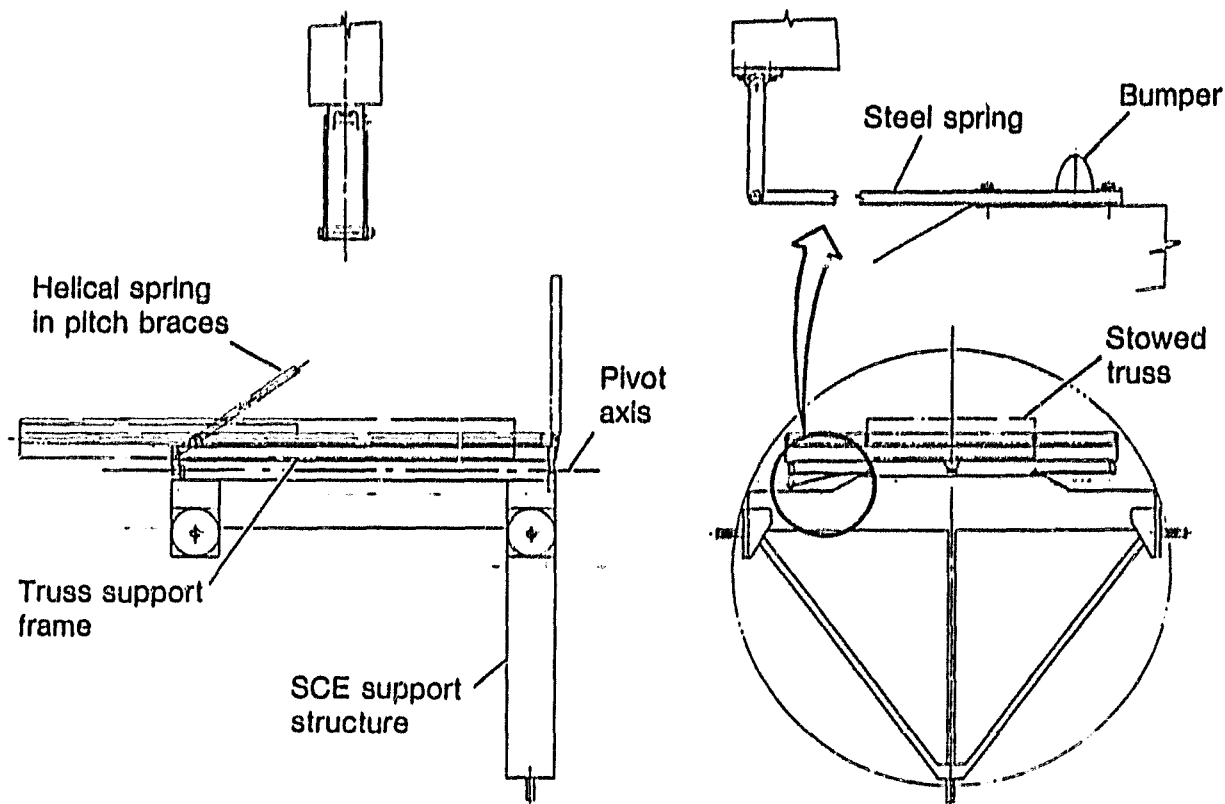
Figure 2-4. Flexible Base Mount Concept

With the truss attached to the Orbiter in a free-free mode, it should be noted that the modal frequencies are higher than they would be if the structure were attached to a fixed support. For example, with the 250 kg tip mass with the rigid base mount, the first mode frequency would be approximately 0.058 Hz as compared to 0.086 Hz with the movable Orbiter.

2.3 FEED MAST RF REFLECTIVITY

For large communications antennas where the desired scan angles are six degrees or less, it appears that the antenna can be designed with an offset feed such that the beam will not intersect the feed mast. This is the most desirable approach because it doesn't require any modification to the selected feed mast structural arrangement. This means the best packaging/deployment arrangement can be achieved for the feed mast. A large offset to the feed system is the simplest approach to pursue. However, this approach can increase the loads that the feed mast must carry and hence the mass of the feed mast somewhat.

For scan angles above six degrees, it may not be effective to increase the feed offset to prevent the beam from intersecting with the feed mast. For this case, several approaches as identified in Table 2-3 would be pursued to meet reflectivity requirements, but each one has major disadvantages. Each of them adds complexity and weight to the structural arrangement of the feed mast. They will degrade the packaging ratio and complicate the deployment operation. They may even change the deployment approach entirely, such as the case where plates are put on the sides of the mast and appropriate cut-outs made in them to make the structure transparent. This arrangement may require a telescopic deployment arrangement, which is less desirable than the selected approach. Additional ground testing would also be required for the feed mast test article to make sure the modifications to the structural arrangement met the requirements.

To allow mast RF reflectivity minimization techniques to be defined and evaluated for incorporation into the flight experiment, further study of specific antenna designs would be required to define requirements for the SCE feed mast test article.

Table 2-3. Feed Mast RF Interference Minimization Approaches

Options	Advantages/Disadvantages	Applications

• Design antenna geometry to avoid interference
  — F/D
  — Offset
  — Frequency
	• Advantages
	  — Does not require modification of selected mast structural arrangement
	  — Best packaging/deployment
	• Disadvantages
	  — More structure/additional mass
		Small scan angles (1 deg - 6 deg)

• Modify mast structural arrangement
  — Provide reflecting geometry
    Metallic reflectors
    Metallized surfaces
    Inflatable reflectors
  — Add absorbing material or coatings
  — Make structure transparent (Dichroic)
	• Disadvantages
	  — Adds complexity & mass
	  — Degrades packaging ratio of mast
	  — Complicates deployment/may change deployment method
	  — Requires additional ground testing
		Large scan angles (>6 deg)

SECTION 3

PRELIMINARY DESIGN AND ANALYSIS

The SCE preliminary design was changed to incorporate the new deployable truss configuration, tip mass jettison capability, flexible base mounting provisions, and the Shuttle Orbiter interfaces. Structural analysis of the deployable truss and its deployment rails and supports was performed and the SCE mass properties were updated. A preliminary Phase 0 Safety Analysis of the SCE payload was also performed. This section presents the results of these design and analysis activities.

3.1 PRELIMINARY DESIGN

The SCEDS Part I baseline concept (reference 1) was selected because it allowed the most latitude in establishing the stowed length of the deployable structure. The general arrangement of the payload would allow it to be integrated with a number of alternative cargo manifests and possibly straddle other palletized experiments such as the Materials Processing Science (MPS) pallet for maximum utilization of cargo space.

The SCE baseline concept arrangement was retained for this phase of the study. The preliminary design was changed as described in the following subsections.

3.1.1 SCE SUPPORT STRUCTURE.
The SCE support structure shown in Figure 3-1 is made up of four open box beams jointed at the ends to form an open rectangular structure. The lateral box beams support the trunnion pin interfaces for the active retention fittings on the Orbiter longeron sill. Within the area bounded by the box beams, additional structure is provided to incorporate an equipment and experiment mounting shelf and a walkway for personnel support required during EVA experiment removal. Located at the c.g. of the SCE support structure, an RMS standard grapple fixture (Spar Part No. 51196F1-3) is provided. This grapple fixture will allow for jettison of the SCE support structure and payload from the cargo bay, using the RMS fitted with a standard end effector.

A spring mounting system for the roll frame assembly is provided in the SCE support structure. Two leaf springs reacting against the roll frame provide a truss mounting stiffness of 1.55×10^5 Nm/rad. To lock the roll frame structure, a sliding pin engages a bracket on both sides of the support structure. A planetary gear motor, through a worm reduction gearing, drives a bell crank which actuates a push-pull linkage to lock and unlock the sliding pins. A shear pin located on the aft portion of the SCE support structure engages an active retention fitting attached to the keel bridge of the Orbiter to react Y-Y loads.

UNLOCK

DETAIL

EXTENSION RAIL (FAR SIDE)

DEPLOYMENT RAIL

EXTENSION RAIL (NEAR SIDE)

UNLOCK

G

F

F

HEX DRIVE FITTING
UNIVERSAL SERVICE
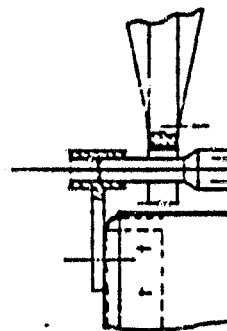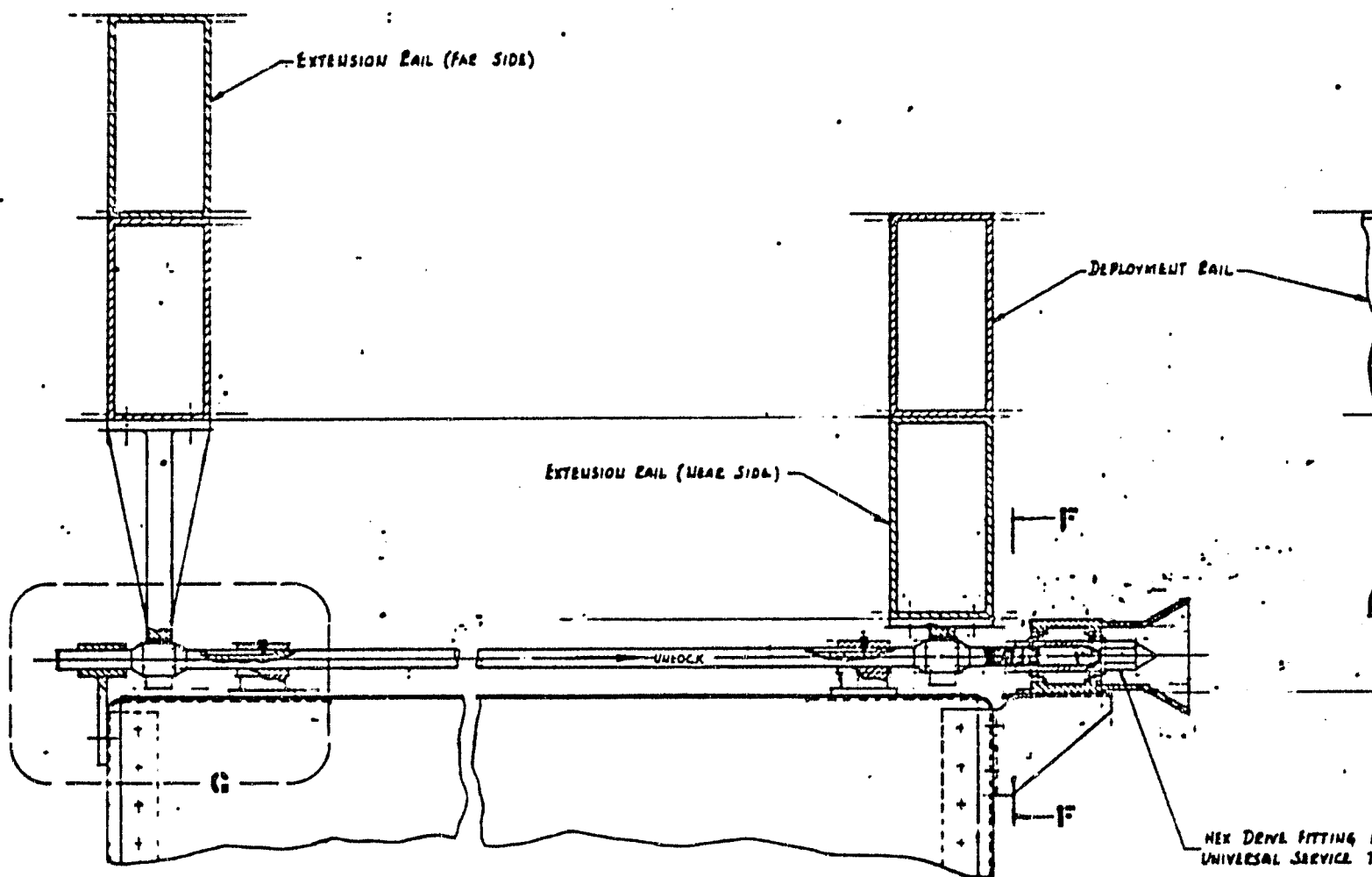
DEPLOYMENT RAIL TO SUPPORT STRUCTURE LOCKING MECHANISM
(SHOWN IN LOCKED POSITION)   SCALE: FULL

SECTION E - E

PLANETARY GEARMOTOR
753:1 SPEED REDUCTION
APPROX 1 MINUTE TO LOCK/UNLOCK
PITCH STRUT
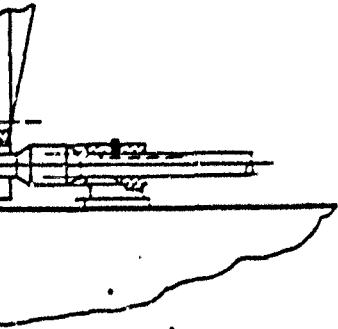
UNLOCKED POSITION

DETAIL G   SCALE FULL

WORM GEAR

SECTION D-D

PLANETARY
REDUCTION.

LOCK

D   D
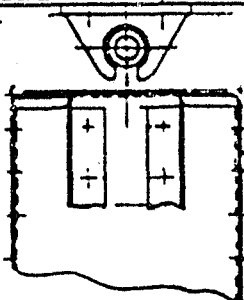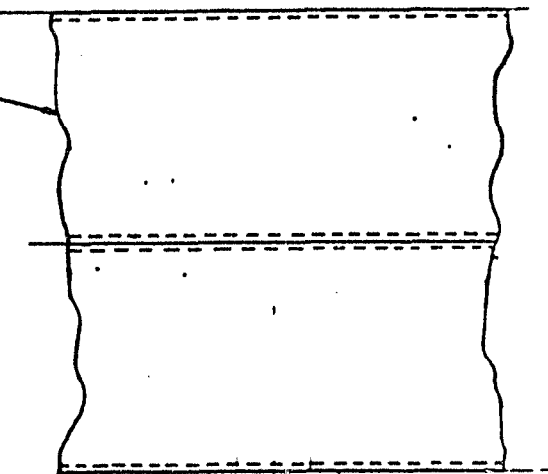
ROLL FRAME STRUCTURE

TING FOR
VICE TOOL

SECTION F-F   SCALE FULL

PITCH STRUT

BALL DETENT LOCK

SPLINE DRIVE

PRE-LOAD ADJUSTING NUT

COMPRESSION SPRING K=112 #/INCH

CARGO

DETAIL 13 SCALE-FULL

WORM GEAR REDUCE 50:1

PLANETARY GEARSYSTEM 7811 SPEED REDUCTION. APPROX 1 RESULT. FOR LOCK/UNLOCK

LEAF SPRING K=1400#/IN

ROLL FRAME LOCKING PIN

D

FOLDOUT FRAME 3

DETAIL C SCALE 1/6

EQUIPMENT MOUNTING SHELF
& WALKWAY

DEPLOYMENT RAIL (4)

TRUSS DEPLOYMENT MECHANISM
(STOWED POSITION)

RMS GRAPPLE FITTING

DEPLOYMENT RAIL
(DEPLOYED POSITION)

TRUSS DEPLOYMENT MECHANISM
(AFT POSITION)

EXTENSION RAIL (FAR SIDE)

PACKAGED TRUSS (30 BAYS @
2.931/BAY = 692.84)

FOLD TRUSS CROSS STRUT

EXTENSION RAIL (NEAR SIDE)

AFT BULKHEAD
X₀ = 1478.28
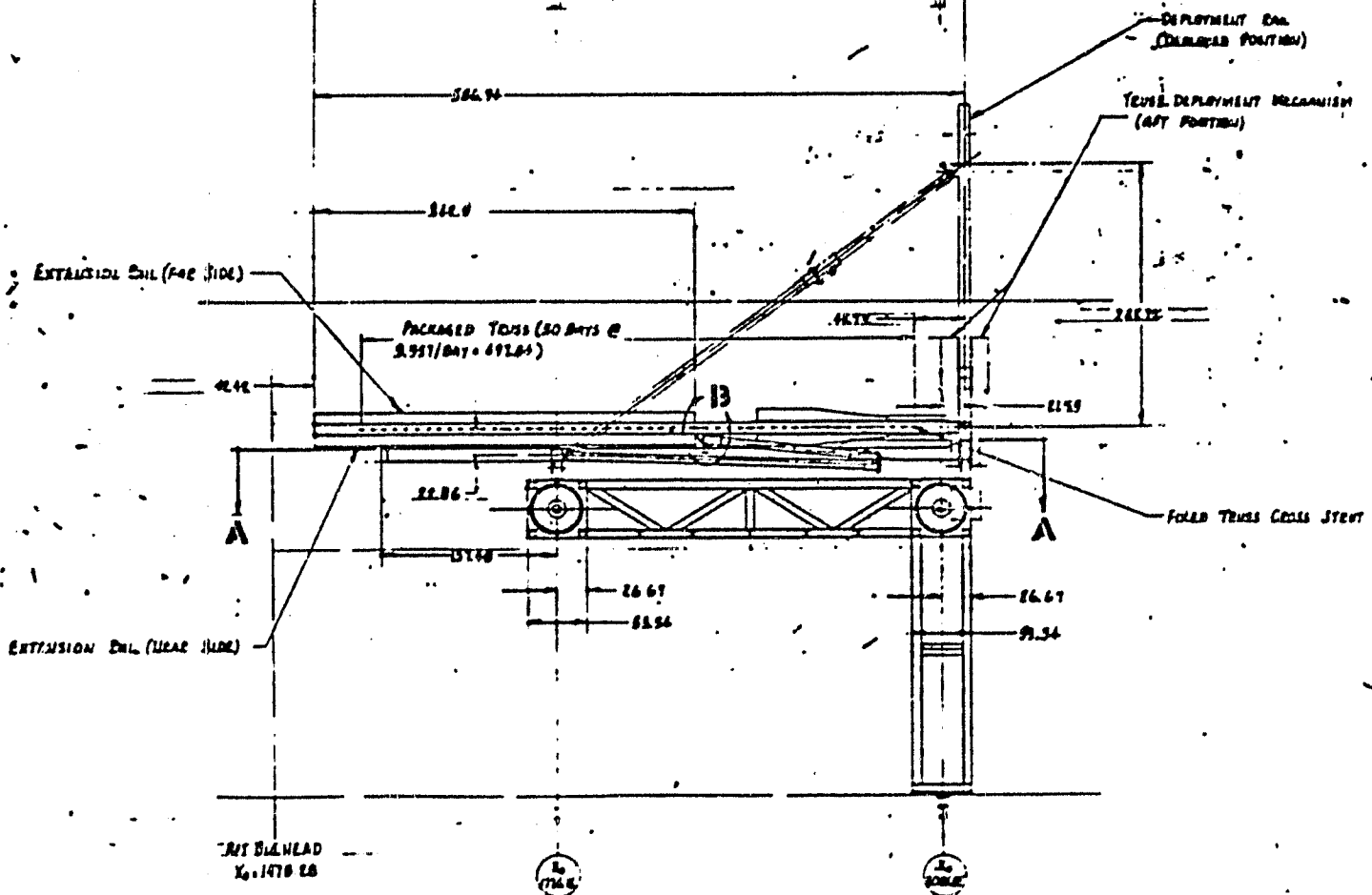
FOLDOUT FRAME 4
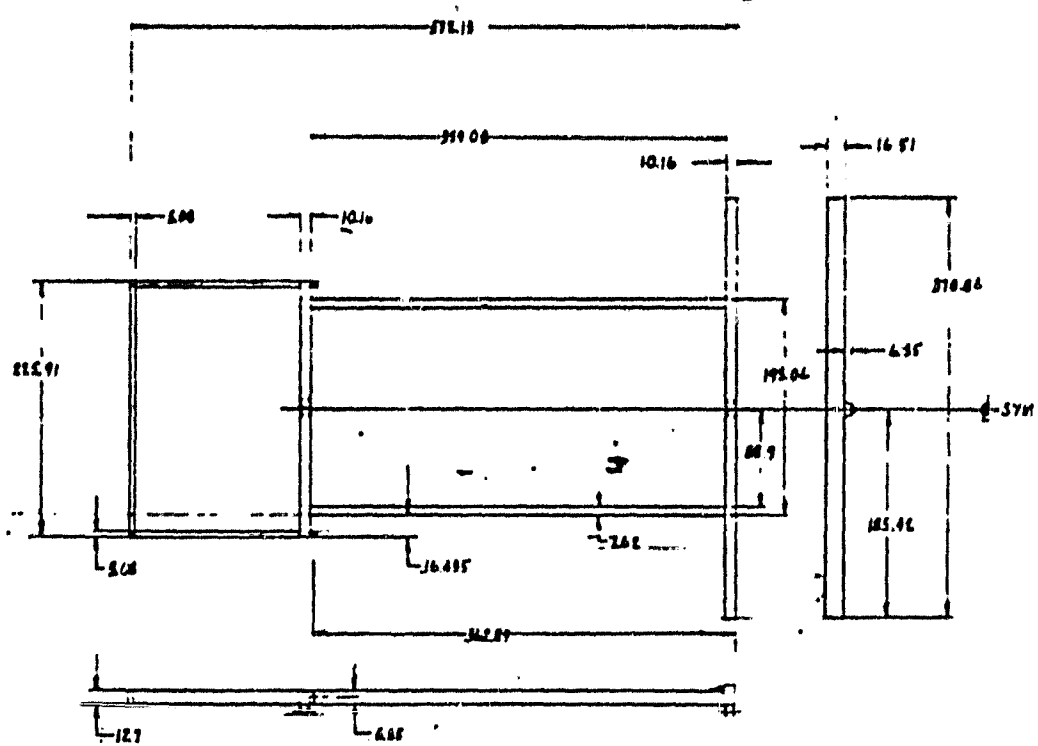
SECTION A-A

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CARGO BAY ENVELOPE
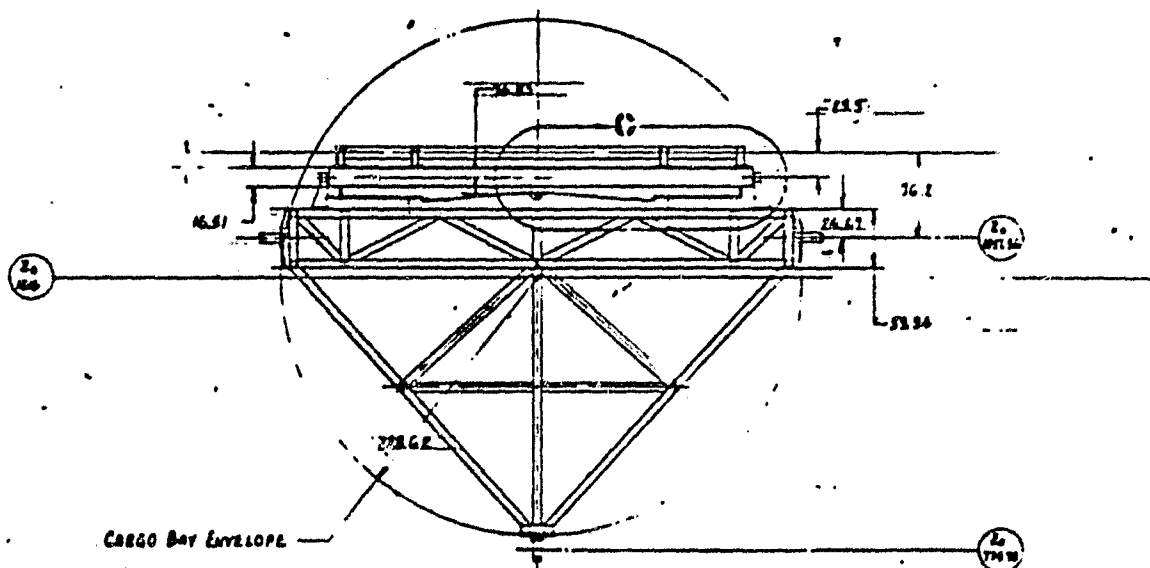
ALL DIMENSIONS IN CENTIMETERS

Figure 3-1. SCE Support Structure

3-2

The roll frame structure is free to rotate in the lateral direction. Mounted to this structure are two truss deployment rails and two pitch struts. Each deployment rail has an extension rail which is rotated 180 degrees for stowage. The deployment rail supports the stowed deployable truss and incorporates the truss deployment mechanism.

Each pitch strut has a spring cartridge that provides a truss mounting stiffness of 1.55×10^5 Nm/rad. Two compression springs (one for each direction of pitch) react against a plunger which is part of the pitch strut mounting bracket. A preload adjusting nut takes out any looseness in the pitch strut at assembly. A ball detent in an annular groove in the adjusting nut locks out the flex mount in the pitch struts. To lock or unlock the strut spring, a planetary gear motor with a splined shaft rotates a threaded taper pin, displacing a spring loaded taper pin which locks or unlocks the ball detents.

The pitch struts contain an overcenter hinge mechanism that locks up when the rail is rotated 90 degrees to the deployment position shown in Figure 3-2. A rotary solenoid is installed on each hinge to allow the struts to be remotely unlocked for restowage.

3.1.2 DEPLOYABLE TRUSS. The general arrangement of the deployable truss is shown in Figure 3-3. The drawing shows the initial stage of the truss deployment with upper and lower lateral struts unfolded and the first two bays deployed. Basically the system consists of a truss deployment rail structure with extension rails, two motorized carriages, two electric cable take up reels, and the deployable truss with a tip-mounted damping augmentation unit and mass. The rails contain tracks for the truss and carriage rollers and gear racks for the carriage drive pinion as shown in Figure 3-4.

To simplify the drives and controls, the RMS is used to perform the following functions: a) rotation of the folded truss/rail assembly; b) deployment and retraction of the two extension rails; c) rotation of the lift and holddown arm for folding and unfolding of the upper and lower lateral struts; and d) rotation of the overcenter hinge tripper support arms.

Linear deployment and retraction of the truss is accomplished by movable carriages. Each carriage contains a drive motor, a solenoid operated latching mechanisms that unlatch the overcenter hinges of the horizontal and upper struts during truss retraction.

The geometry for one folded truss bay is as drawn in Figure 3-4. All strut tubes are of the same diameter. Clearances between struts and hinges are held to a minimum to obtain the smallest package possible. A trip arm is attached to each overcenter hinge. This arm is used to initiate the folding sequence of the overcenter hinge struts during truss retraction.

.9.14

SPLINE
DRIVE

8.13
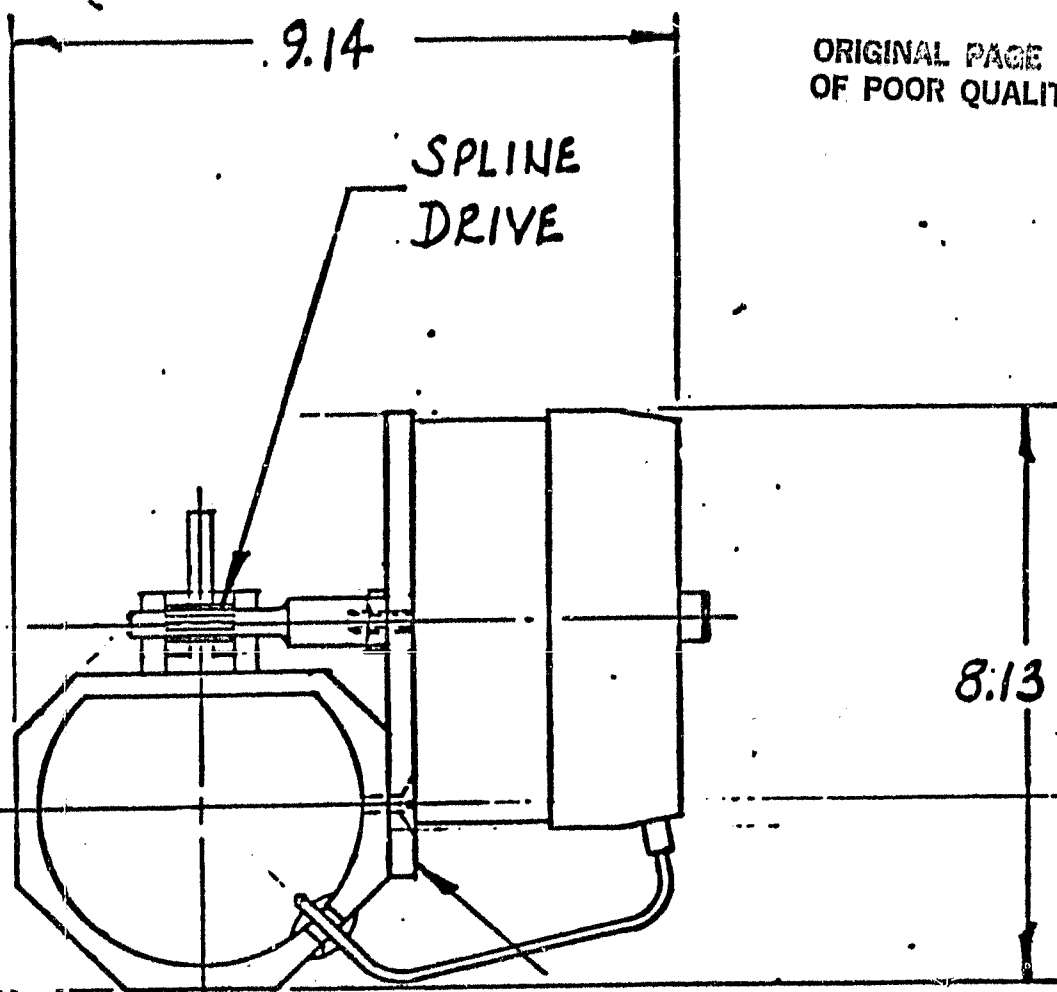
SOLENOID MOUNTING
BRACKET

SECTION A — A

A

A

3.05

13.21

NG
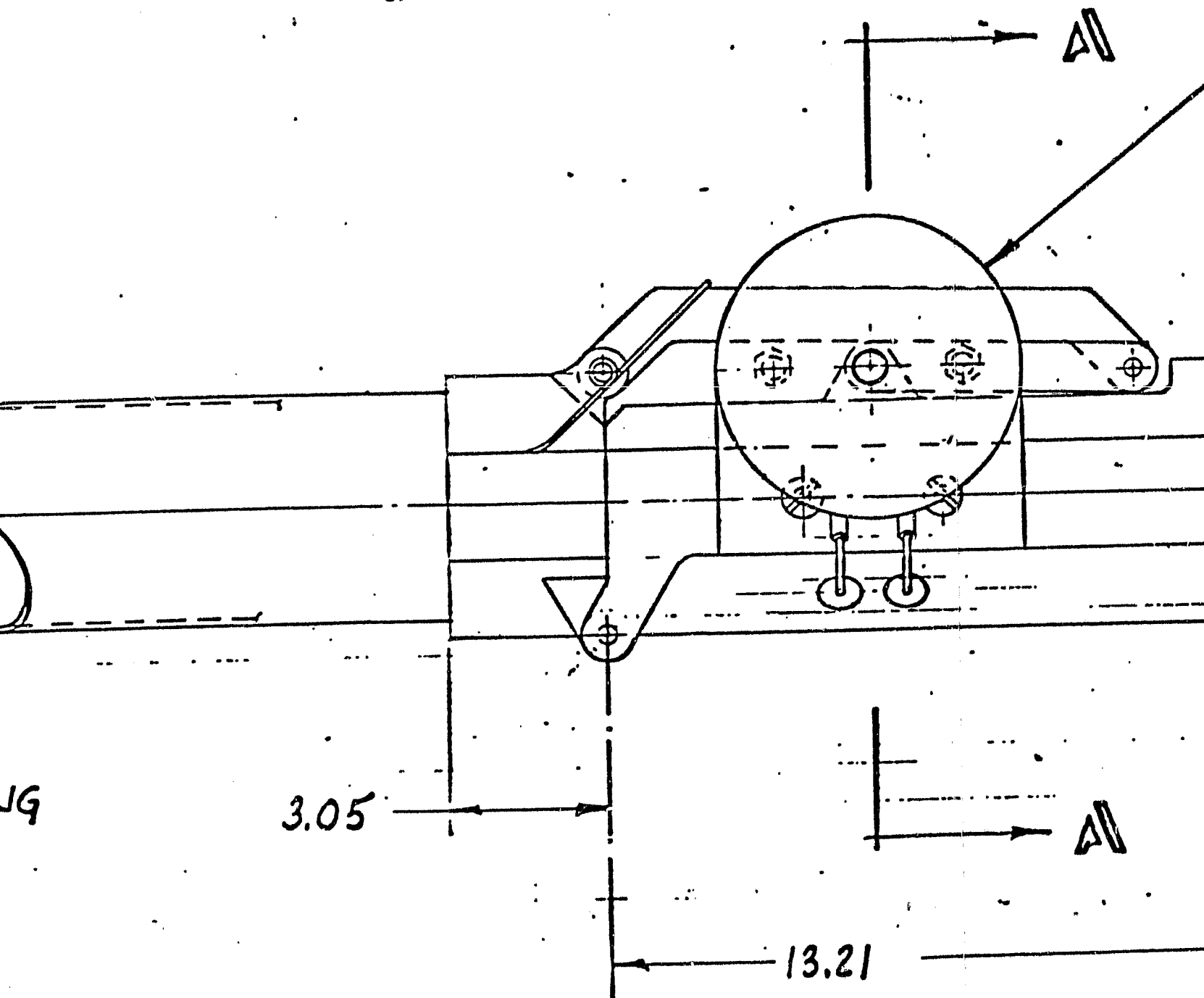
LEDEX ROTARY SOLENOID - SIZE 6S
2.17 KG-CM TORQUE AT 45°
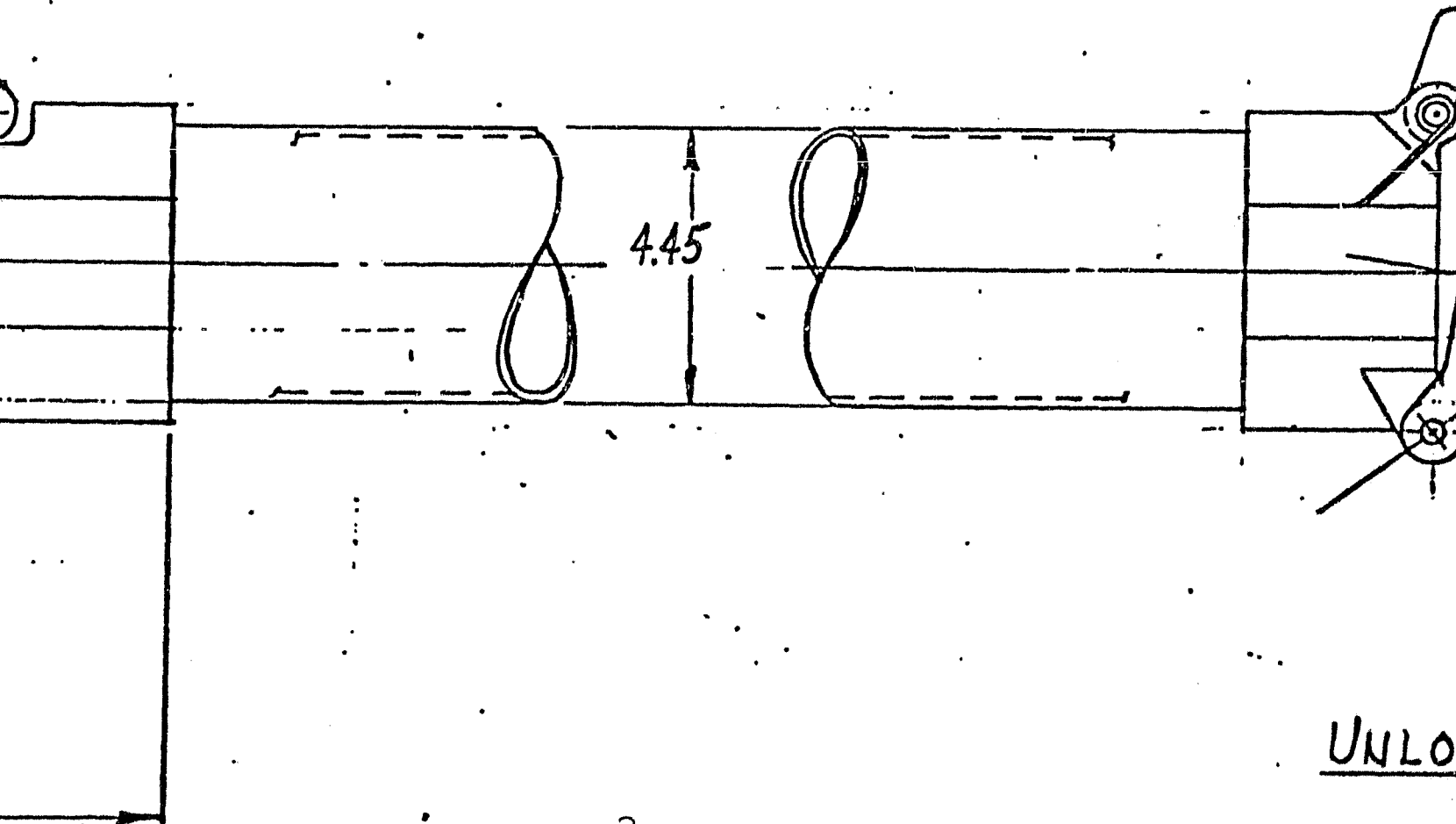
4.45

UNLO

45° STROKE
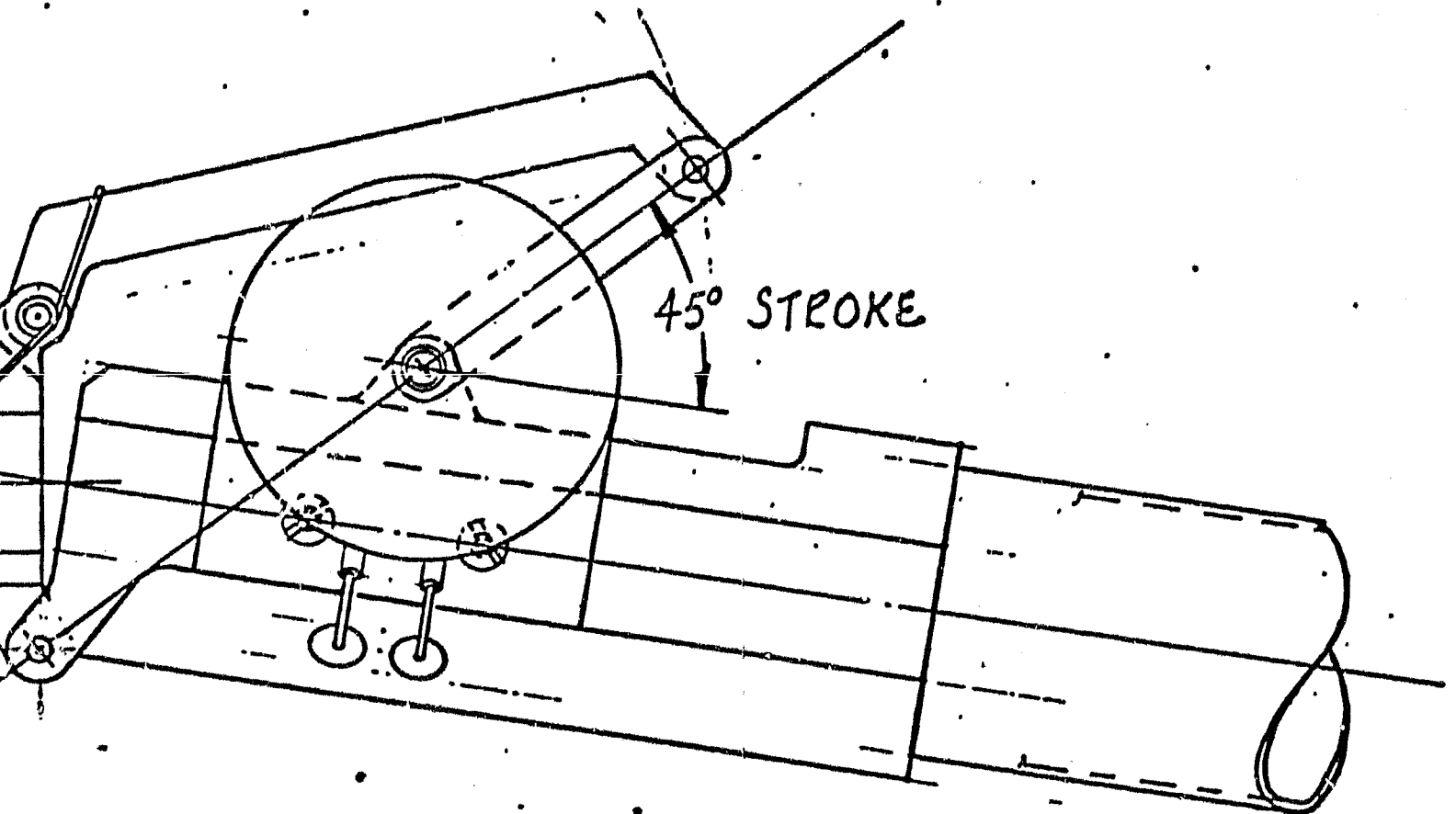
LOCKED POSITION

ALL DIMENSIONS · IN CENTIMETERS

FOLDOUT FRAME 5

Figure 3-2. Pitch Strut Over-Center Hinge Release Mechanism

TRUSS EXTENSION RAIL

EX

DAMPING AUGMENTATION UNIT

200

200

225.9

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A

TRUSS FOI

OVERCENTER HINGE TRIP A

CARRIAGE (UPPER)

DAMPING AUGMENTATION UNIT

A

283

TRUSS EXTENSION RAIL

EXTENSION RAIL RETRACTION DRIVE RECEPTACLE FOR RMS

R...
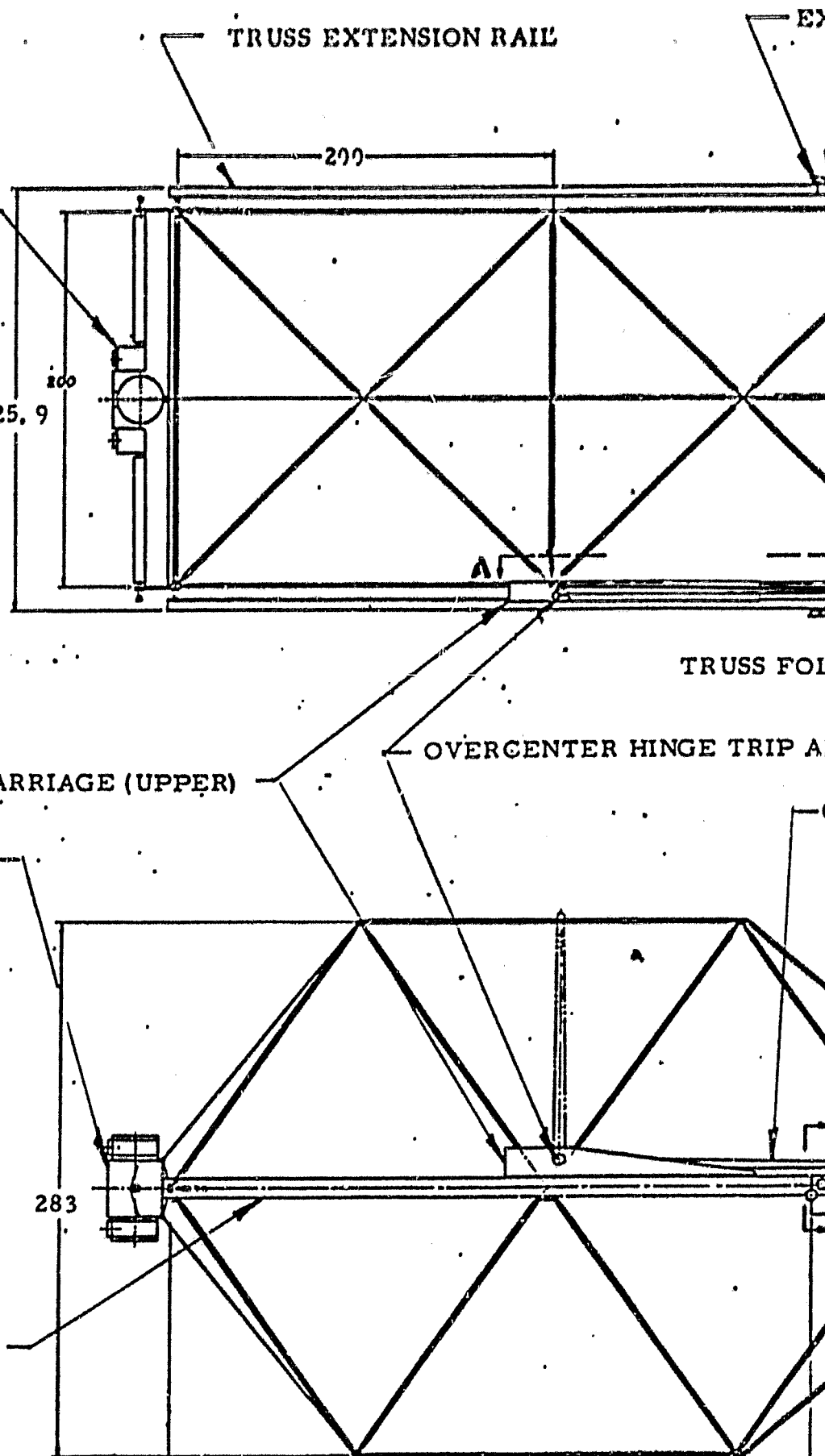
EXTENSION RAIL DEPLOYMENT DRIVE RECEPTACLE FOR RMS

TRUSS DEPLOYMENT RAIL

358

CARPENTER HINGE TRIP RAIL (STOWED)

...A

...OLD DRIVE RECEPTACLE FOR RMS

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...P ARM ROTATION RECEPTACLE FOR RMS

OVERCENTER HINGE TRIP ARM (STOWED)

CARPENTER HINGE T...

TRUSS FOLD DRIVE RECEPTACLE FOR RMS

ELECT...

H

B

H

B

346.6

TRUSS DEPLOYMENT RAIL

RAIL

EXTENSION RAIL RETRACTION DRIVE
RECEPTACLE FOR RMS

FOLDOUT FRAME

579.1

RAIL PIVOT AXIS

R RMS

358

ELECTRIC CABLE TAKE-UP REEL

Y

CARPENTER HINGE TRIP RAIL

ELECTRIC CABLE TAKE-UP REEL

S

346. 6

FOLDOUT FRAME

RAIL PIVOT

DIMENSIONS IN CENTIMETERS

FRAME

Figure 3-3. Deployable Truss General Arrangement

OVER-CENTER HINGE TRIP ARM

CARPENTER HINGE TRIP RAIL

CARPENTER HINGE TRIP ARM

CARP

G - G

CARPENTER HING

OVER-CENTER HINGE
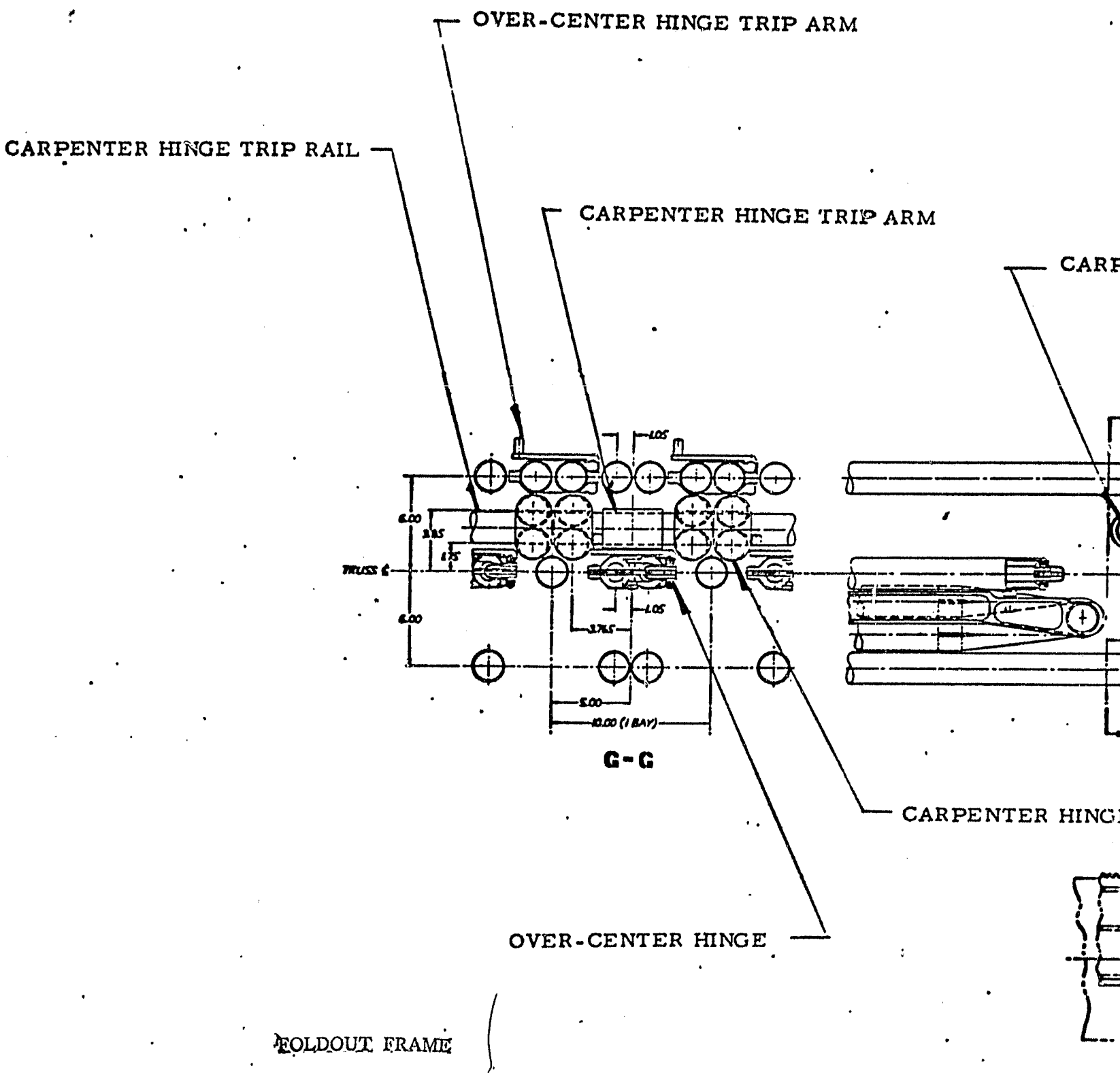
DEPLOYMENT RAIL

CARPENTER TAPE HINGE

H-H
SH I

EXTENSION RAIL

G

G

B-B
SH I

100.00

HINGE STRUT

F

RETRACT

FOLDOUT FRAME 2

DEPLOYMENT RAIL

F

OVER-CENTER LATCH

RETRACTION SHAFT

FOLDOUT FRAME.

TRIP LEVER

LIFT & HOLD-DOWN ARM

DEPLOY

OVER-CENTER HINGE

RETRACT

DEPLOYMENT SHAFT

WORM GEAR

ROTARY SOLE

DEPLOY

FIXED NODE

DRIVE

45°

8.50

10.00

1.40

6 FIXED STRUT

45.70

100.00

E

C

HINGE TRIPPER

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TRIPPER SUPPORT
ARM

CARRIAGE

4

FOLDOUT FRAME

E-E

LENOID

F-F

D

CARRIAGE DRIVE MOTOR

Figure 3-4.

E LATCH

GEAR RACK

G-G

D-D

RIPPER SUPPORT
ARM

CARRIAGE

DEPLOY/RETRACT
SHAFT

E-E

FOLDOUT FRAME

DIMENSIONS IN CENTIMETERS

Figure 3-4. Deployable Truss Mechanism

3-6

A cross-section of the carpenter hinge trip arm is also shown. Rails attached to this arm hold the carpenter hinge struts in the folded position. The arm is rotated by RMS Universal Service Tool drive action to deploy the lateral members. For retraction of the lateral members, the trip arm collapses the carpenter tape hinges and folds the lateral members to the stowed position.

The end of the deployable truss is equipped with a special support frame for the damper sets and tip mass (Figure 3-5). Six damper sets, each consisting of a torque motor, rotor, and rate gyro sensor are mounted in a housing such that there are two damper sets per axis. The effects of varying the damping ratio from 0 to 1% to 2% along each axis can thus be evaluated.
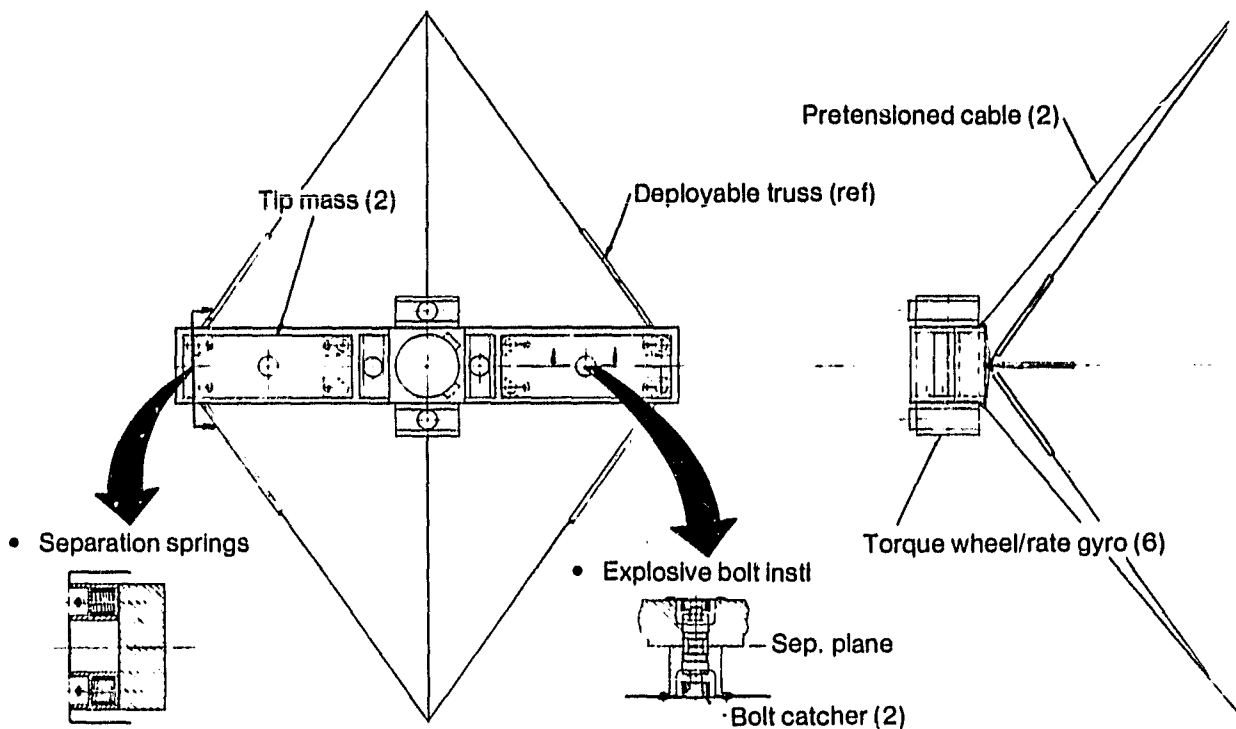
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Figure 3-5. Damping, Excitation and Tip Mass Assembly

Two steel bars are attached to the support frame, each by an explosive bolt. The steel bars provide the added mass necessary to bring the total weight of the tip package to 250 kg. However, the tip masses must be jettisoned to provide a favorable center of gravity of the experiment for payload jettison in the event of a retraction failure of the truss. The tip masses are jettisoned by firing explosive bolts, allowing separation springs to accomplish the jettison.

The support frame is supported at the center by two pre-tensioned cables attached to the truss. These cables react the moment loads that will be generated by the torque wheels in the damper sets during damping or excitation operations. These cables deploy and stow with the aid of the carpenter hinge trip rail.

3.1.3 SCE CONTROLS. The hardwire control concept defined in SCEDS Part I was updated as shown in Figure 3-6 to incorporate the following changes:

a. The Control Unit (CU) was relocated from the Aft Flight Deck (AFD) to the SCE support structure in the cargo bay. This change provides the following advantages:

  1. Hardwire interfaces are reduced from 194 to 54 and wire lengths between the CU and the truss are reduced.

  2. Cargo bay 28 VDC power is used. This provides a larger power source than the AFD (0.35 vs 7 kW).

  3. CU does not have to be qualified to AFD offgassing and heating requirements.

b. The dedicated payload AFD control panel was eliminated. The Orbiter-provided standard switch panel (SSP) will be used to control the operation of the SCE. The Orbiter-provided active latch controls will be used for jettison control and the Orbiter-provided cathode ray tube (CRT) display and keyboard will be used for monitoring status and data.

c. Dual PCM outputs provided direct payload data interlevel (PDI) and payload recorder interfaces. This will allow data to be recorded and/or downlinked in real time.

d. Mechanical controls for caging and release of the flexible base mount springs and unlatching pitch strut hinges were added.

e. Instrumentation was updated and separate units for PCM encoder and signal conditioner were defined.

The SCE CU concept is shown in Figure 3-7. The unit is a modified version of the CU being developed for the Shuttle/Centaur high energy upper stage. The unit has the following characteristics:

a. Weight: $\sim$ 20 kg (maximum capacity)
            $\sim$ 15 kg (for SCE)

b. Power: 45W

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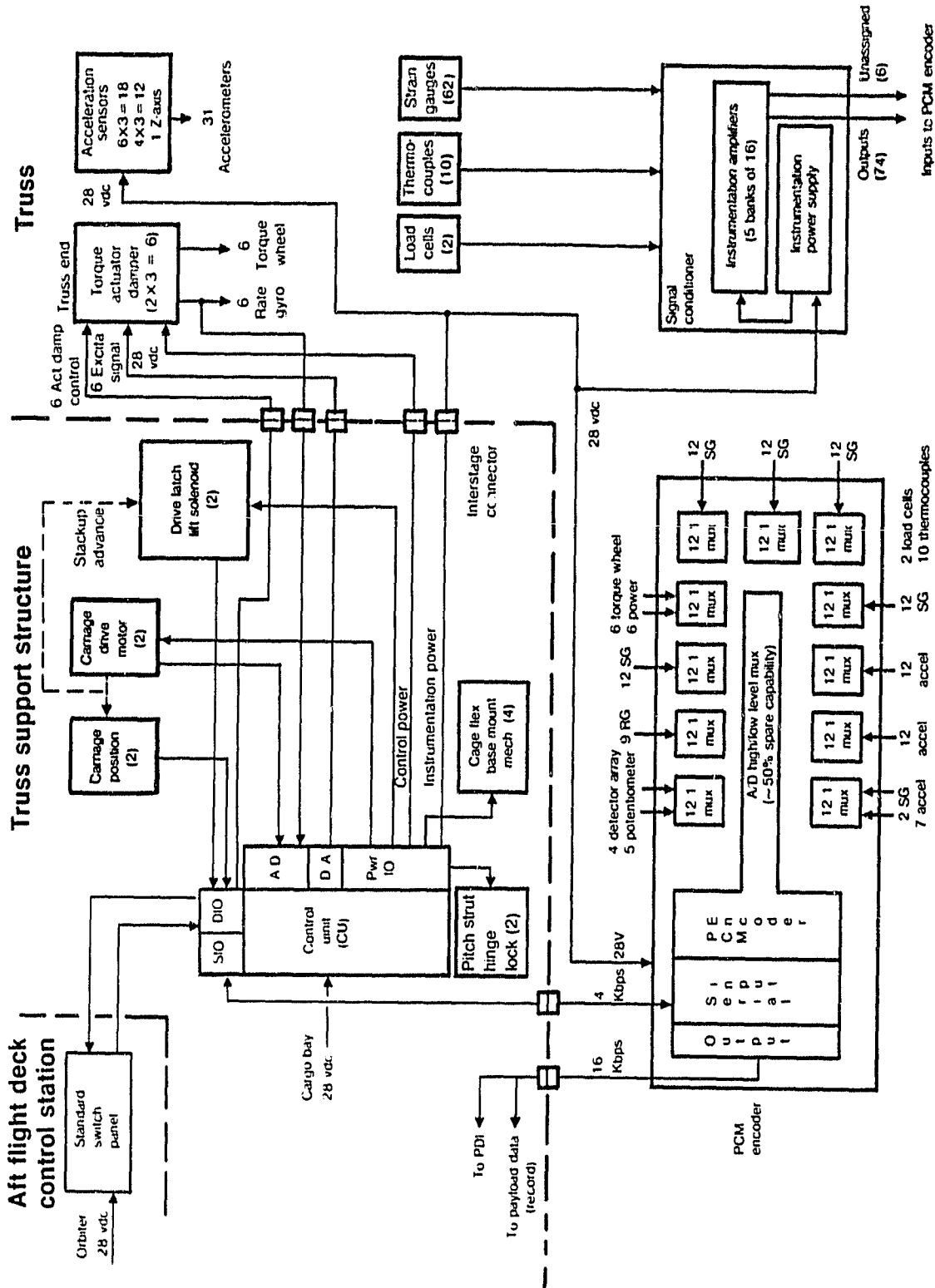
Figure 3-6. SCE Updated Control and Instrumentation Concept

Major software functions
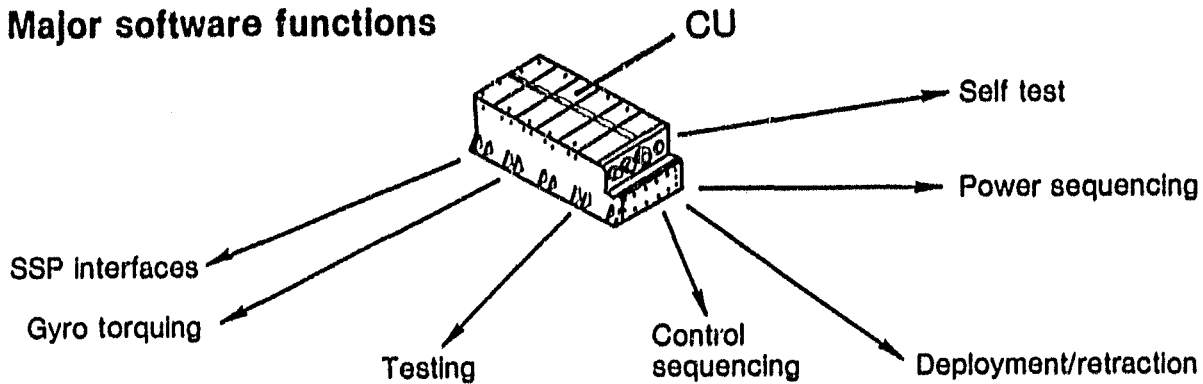
Figure 3-7. SCE Control Unit Concept   17121784-3

c. Inputs: 16 A/D, 45 discretes

d. Outputs: electromechanical or solid state relay

e. Control unit: Z80 microprocessor, 8k of 8-bit PROM memory,

f. 1k of RAM memory

g. CCLS compatible (Centaur Computer Controlled Launch Set available at KSC and Factory).

h. Series inhibits in the power control section of safing functions.

The SCE CU software is defined as that software which executes within the SCE Control Unit. This software is used to support:

a. Ground testing at GDC, KSC, the VPF, and Cx39.

b. Ground and launch support operations.

c. Predeployment testing and operations of the experiment.

d. Deployment, retraction, and safing operations.

Based on the functions identified for the SCE, and assuming electronic computattion of deployment motors and damper torque motors, the estimated memory required is approximately 1700 locations of PROM and 400 locations in RAM. The CU memory capacity provides adequate margin for SCE requirements.

3.2 SHUTTLE ORBITER/SCE INTERFACES

The interfaces and interface hardware requirements for the SCE integrated with the Space Shuttle system were identified and defined. The power, data, control, and mechanical interfaces, as well as Orbiter provided capabilities to support the basic flight experiment, are described in the following subsections.

3.2.1 POWER, DATA AND CONTROL INTERFACES. The following power, data, and control interfaces have been identified as shown in Figures 3-8 and 3-9. They are summarized as follows:

a. Telemetry and Data Services

    1. Interfaced at the Standard Mix Cargo Harness (SMCH), originating at $X_0 603$ (inches).

    2. PCM interfaced to P/L Data Interleaver (PDI) and P/L recorder.

    3. PDI provides real-time data to ground.

    4. P/L recorder enables additional recording and playback.

b. Displays and Controls

    1. Interfaced at SMCH, originating at $X_0 603$ (inches).

    2. Caution and warning (if required) - hardwired interface.

    3. PDI interface enables display of SCE PCM data on CRT.

    4. Retention control system for Orbiter supplied latches.

    5. Standard switch panel - switch commands and talkbacks.

c. Power Interfaces

    1. Utilize DC power from cargo bay at $X_0 645$ (7kW available to be shared).

    2. If available use, AFD DC power (0.35kW).

A standard connector panel hard-mounted to a mid-fuselage frame, as shown in Figure 3-10, will be used for the payload interfaces for power, data, and control harness connections. The orbiter harnesses connect to this panel. The SCE payload includes a set of lanyard pull-type connectors designed to separate if connectors are a flight-qualified low-cost configuration used on Atlas launch vehicles for staging separation of flight harnesses.

The SCE control philosophy is based on the following ground rules and assumptions:

a. The CU will be operated only during pre-launch checkout (C/O) and during experiment.

b. All SCE interfaces will be checked prior to launch.
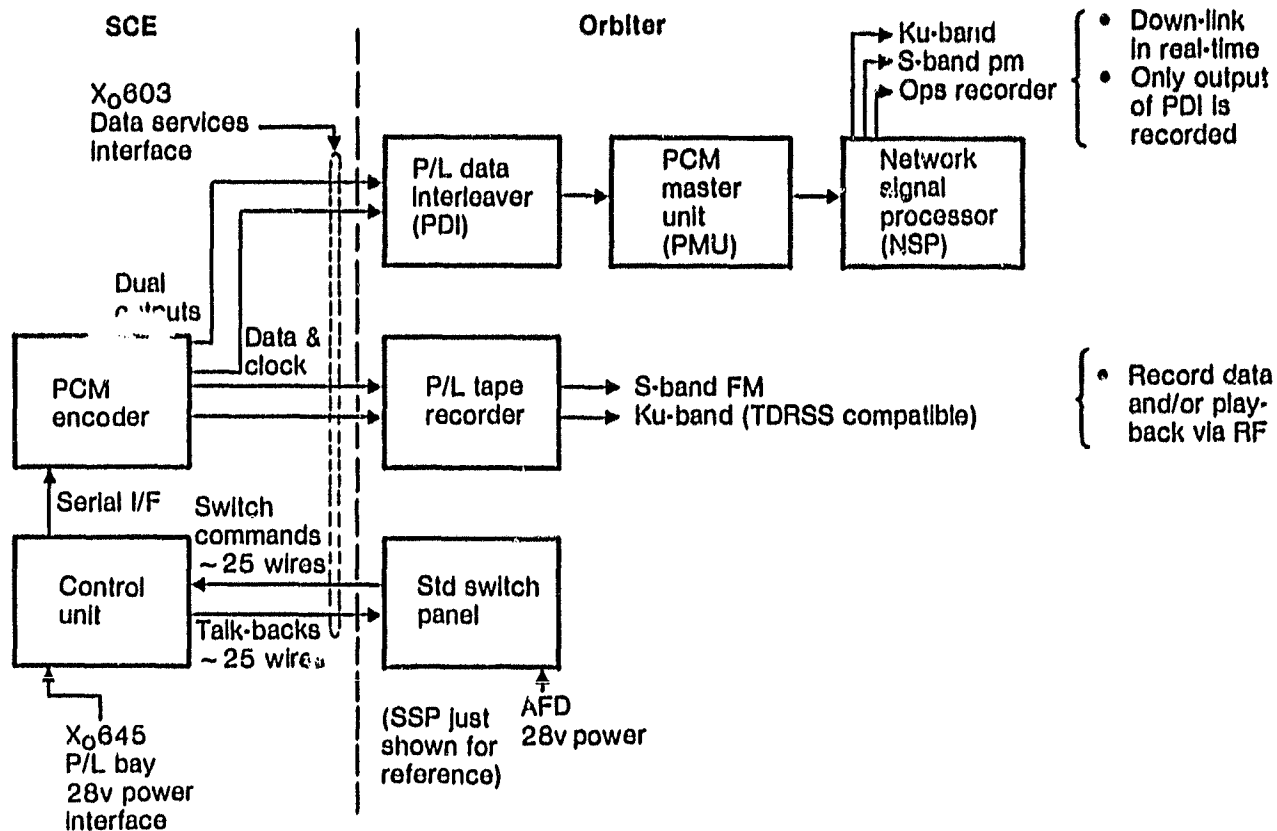
Figure 3-8. SCE Telemetry and Data Interfaces

C&W status inputs
- Direct hardwired measurements
- Control & status from CU

SCE PCM data
- Single input channel
- Only used during SCE operations

Retention control panel
- On-orbit active sill & keel latches
- For jettison of SCE via orbiter latches (No direct SCE avionics interfaces)
- Control & status of latches

Standard switch panel (P/L specialist station)
- SCE power commands
- Deployment/retraction commands
- System status talk back signals (28v discretes)

Aft flight deck CRT displays
- Displays data from SCE via PDI:
  - Deployment mechanism carriage & drive latch
  - Structure gyro and acceleration sensors
  - Status and redlines

Figure 3-9. Display and Controls Summary
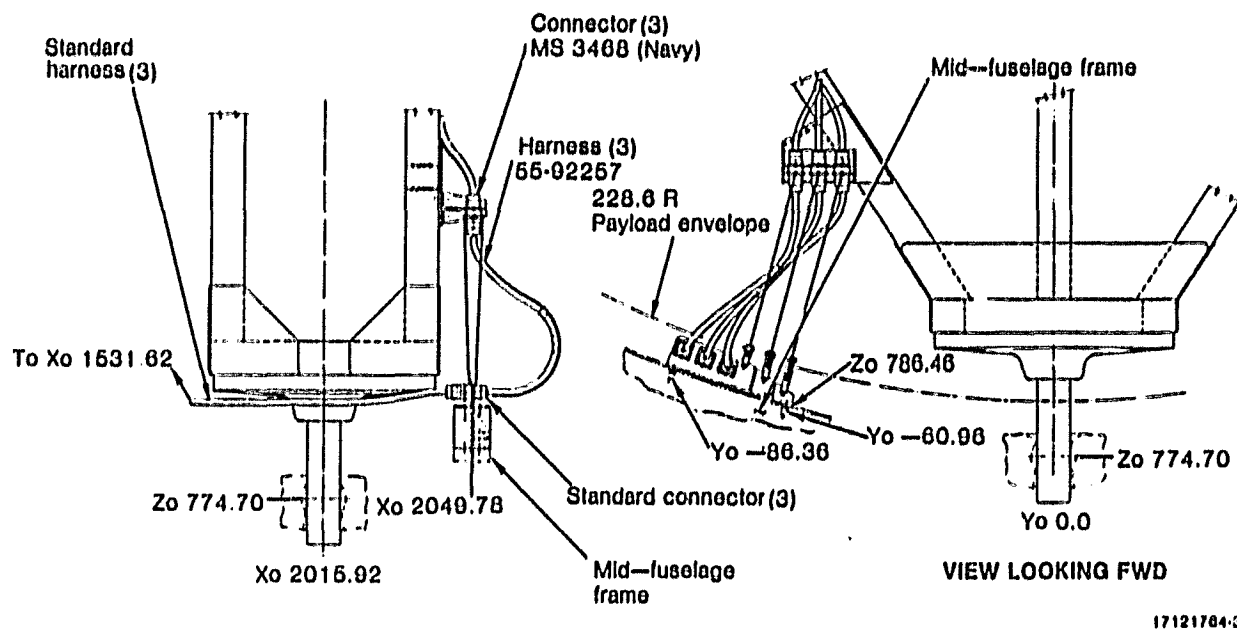
Figure 3-10. Payload Power, Data and Control Harness Interface

c. The SSP is the safing function for the CU. This includes master power
    kill and the ability to enable safety-critical functions.

d. The CRT displays experiment progress data from the instrumentation and
    the CU.

e. Jettison is last resort if retraction fails.

f. Payload jettison is controlled by Orbiter provided functions.

The fundamental concept behind the control sequence is that the SSP is used in
a step-by-step manner to initiate individual preprogrammed CU functions.
Continuation to the next step is justified by the checkout and priming necessary
at that phase in the sequence as illustrated in Figure 3-11.

Failures will be assessed by the crew, with retraction being the first backout
procedure. If retraction becomes impossible, the final safing mode will be to
jettison the experiment.

The basic SSP functions listed in Table 3-1 allow for a simple interface to the CU
while providing flexibility in the type and number of control sequences the CU
can perform. The SSP also acts as a series inhibit safing function for operations
also controlled by the CU.

In addition to the sequence and status indications on the SSP, further insight
into experiment operation can be extracted from PCM instrumentation data and
displayed on the CRT as summarized in Table 3-2. Preprogrammed CRT page
formats can be called up for display.

ECS: Erection Control Sequence

RCS: Retraction Control Sequence

Jettison: Connect RMS
release tip mass
release orbiter latches
extract truss

CU avionic c/o — (self test)
- A/D converter
- Discrete Input Section
- Microprocessor Control Section

CU subsystem c/o
- Verification tests &
  health monitor
- Reasonableness tests
  on subsystem I/Fs
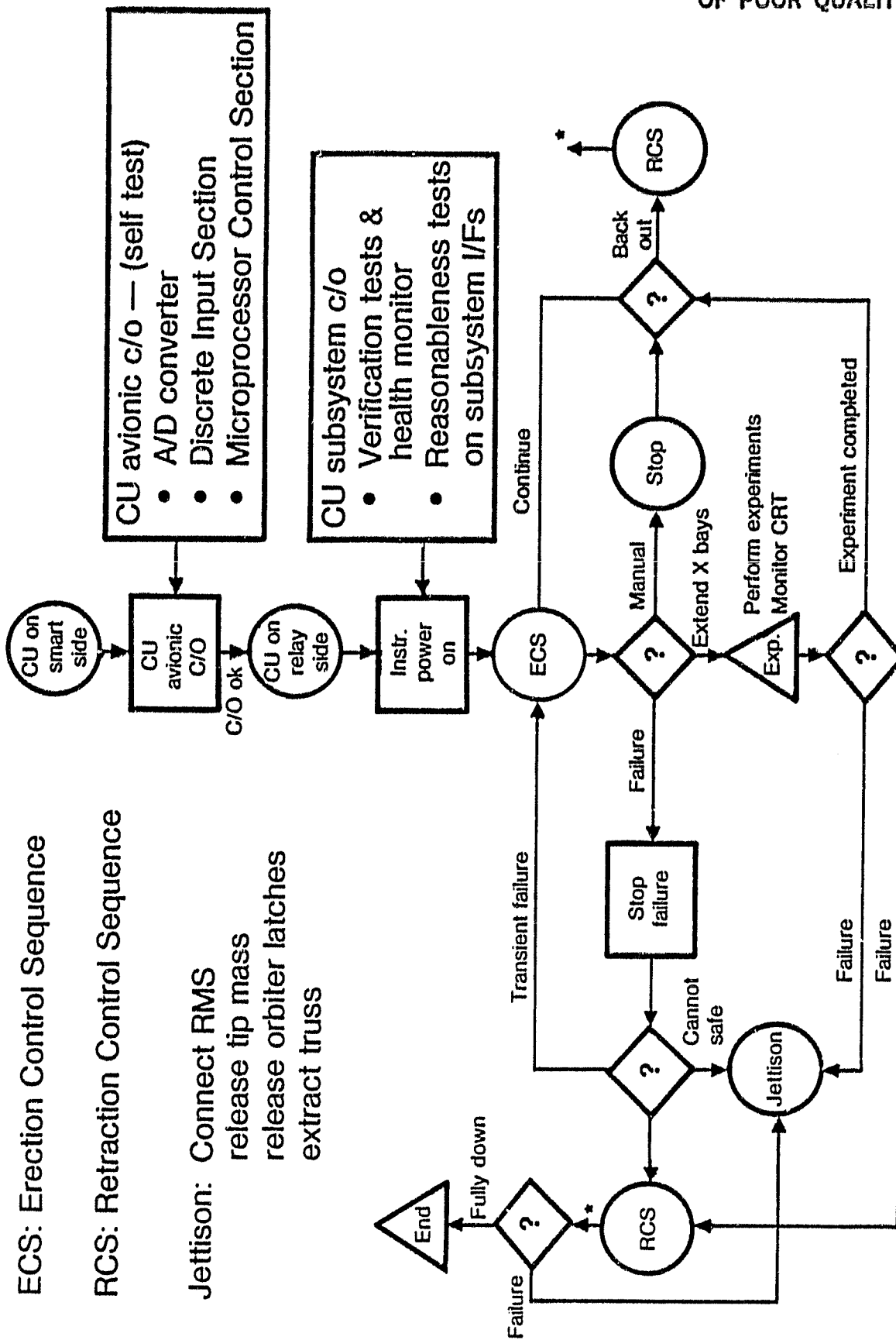
Figure 3-11. SCE On-Orbit Control Sequence

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Table 3-1. Standard Switch Panel Functions

SSP Status Indicators
- Truss extension (1/3, 2/3, full)
- Stopped
- Retracting-fully down
- Smart side power on
- Relay side power on
- C/O ok - start up sequence completed
- Safety arm status

SSP Initiation Control Functions
- Power on/off - CU smart side and initiates avionics c/o
- Power on/off CU relay side and instrumentation
- Erection Control Sequences (ECS)
- Stop
- Retraction Control Sequence (RCS)
- Tip mass safety jettison arm

Table 3-2. CRT Panel Control and Display Functions

Call Up Experiment Data/Status
- Deployment mechanism carriage and drive latch
- Structure gyro and acceleration sensors
- Control Unit digital readout
- Status and redlines
  - CU
  - Structure
  - Power
  - Safety-tip mass
  - Orbiter latch status

Caution and Warning Philosophy
- CU issues status discretes of subsystems and CU health
- CRT and keyboard used to identify/isolate failures

An in-house developed caution and warning procedure based on other Shuttle
integration studies, such as Shuttle/Centaur, is easily implemented for SCE.
The CU issues self test signals and evaluates system health. Upon receipt of
a caution and/or warning signal, the CRT keyboard is then used to call up
instrumentation on a malfunctioning subsystem and evaluate the next or alternate
procedure to be followed.

During the experiment two crew members must work in the AFD to initiate and monitor SCE operations and to operate the RMS. Figure 3-12 shows the physical relationships of the CRT, payload retention panel, RMS panel and the probable location of the SSP.
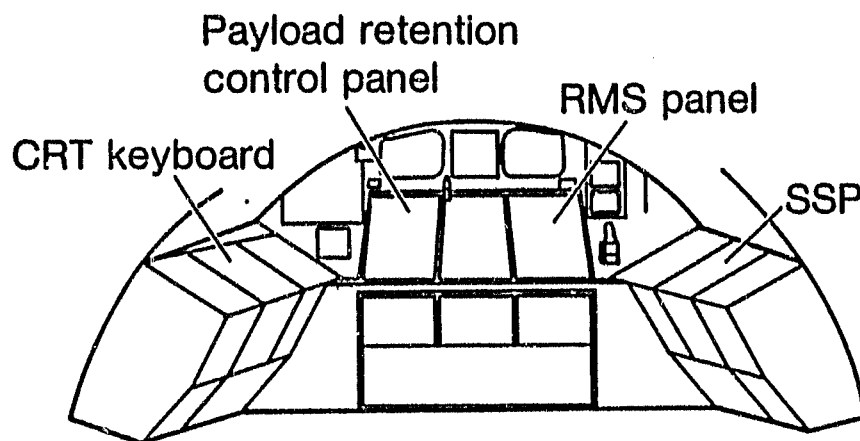
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Figure 3-12. Aft Flight Deck Control Station

On-orbit operations require the use of the SSP for deployment/retraction (function initiation). The retention control panel is used to control those Orbiter latches which enable the jettison of the experiment during safing procedures. The CRT is used for display of back-up information for both the SSP and retention control panel functions.

The payload specialist may opt to move from the CRT to the SSP to perform experiment switching functions or may request the RMS operator to perform the switch actuation function when the procedure requires it. Since the switching functions are not time critical, the most convenient method may be used. Panels may also be relocated to some extent if necessary to enhance ease of operation.

3.2.2 MECHANICAL INTERFACES. The SCE support structure employs a standard five point payload retention system with four longeron attachments and one keel fitting attachment. Active longeron and keel fittings are Orbiter provided to allow jettison of the payload with the RMS. Standard Y-guide plates and scuff plates are installed to ensure proper clearance between the payload and the Orbiter structure during jettison as shown in Figure 3-13. The keel fitting interface between the Orbiter active keel fitting and the SCE support structure is detailed in Figure 3-14.
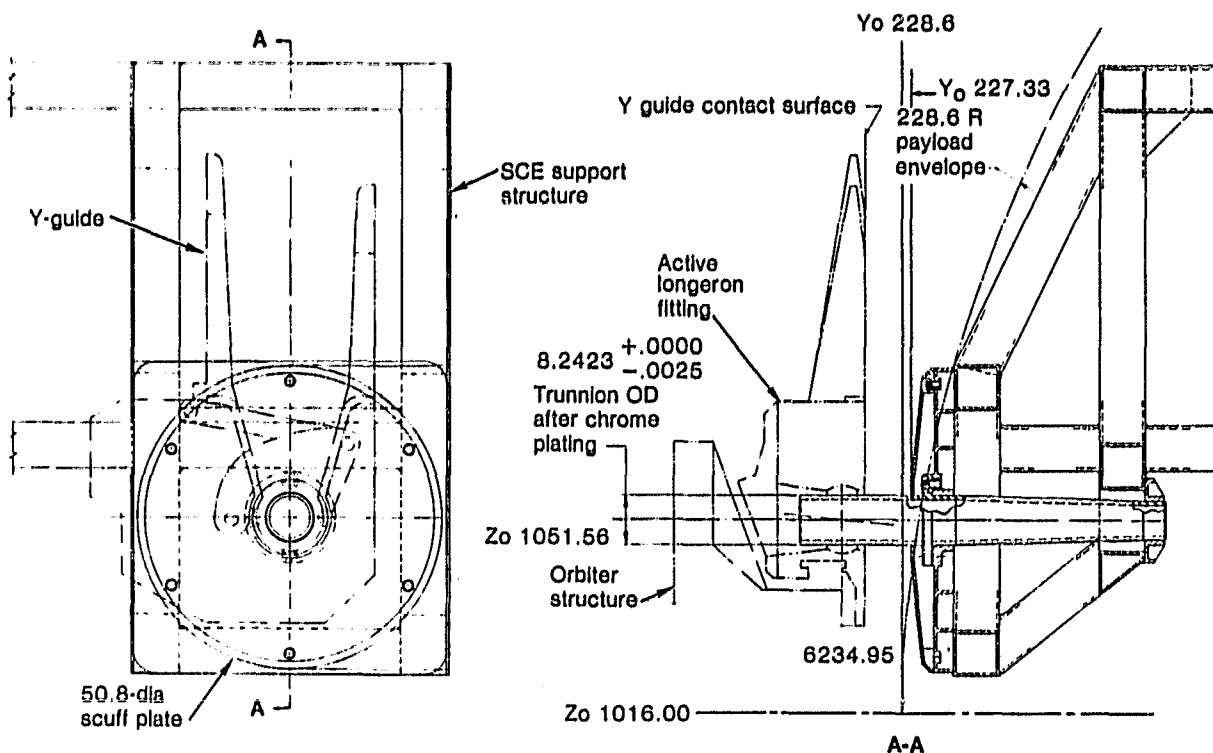
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Figure 3-13. SCE Payload Retention Longeron Attachment - Typical 4 Places
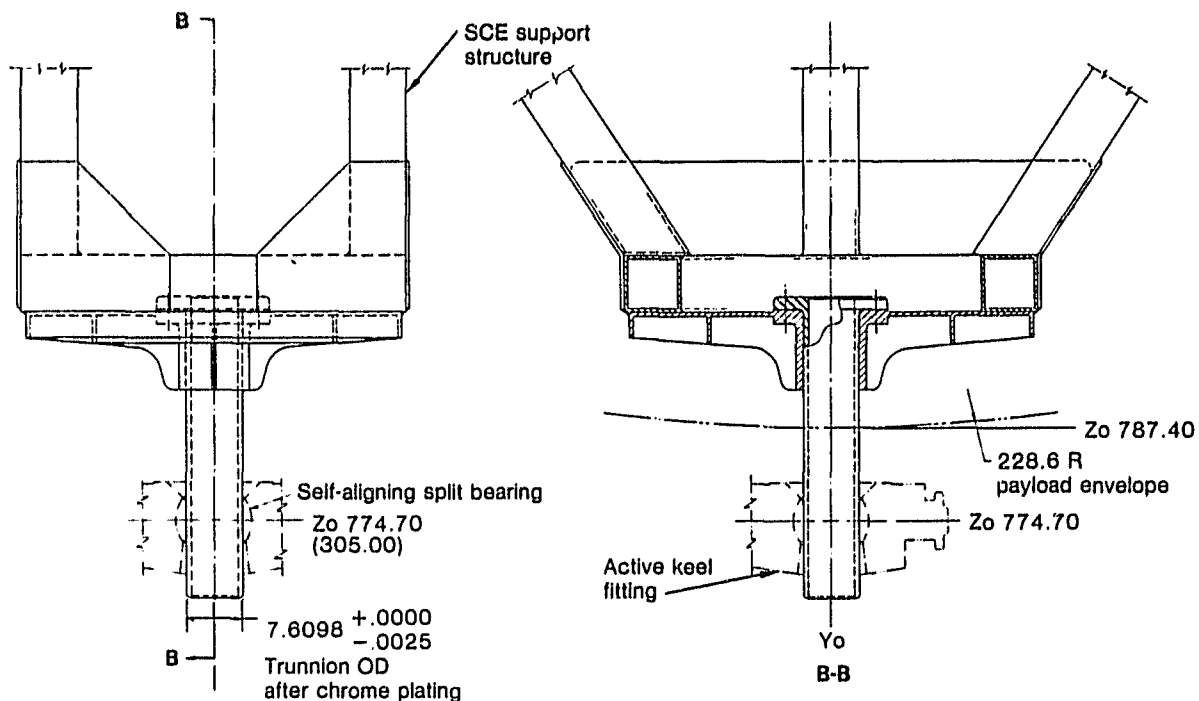
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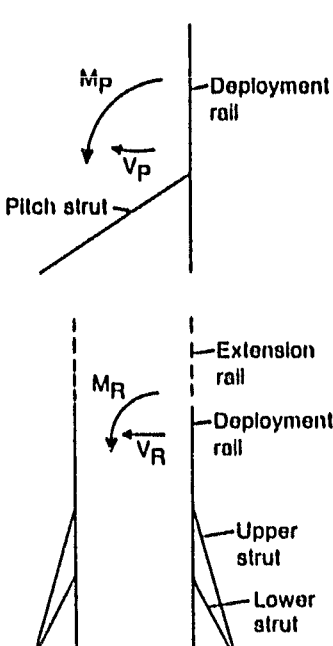
Figure 3-14. SCE Payload Retention Keel Fitting

3-17

3.3 DESIGN ANALYSIS

Analyses were performed to verify the structural capability of the revised SCE truss and truss support structures. Mass properties were also updated to incorporate the latest configuration data.

3.3.1 **STRUCTURAL ANALYSIS**. Deployment rail loads were computed for the new deployable truss configuration with a 250 kg tip mass. Shear and moment loads applied in pitch and roll were determined for the VRCS thrusters "on" case. The maximum loads summarized in Table 3-3 vary with deployed length due to the relative positions of the truss deployment support rollers in the rails. These maximum loads were determined to be well within the structural capability of the rails and allowed the cross-sectional size of the aluminum rail support struts to be substantially reduced.

Table 3-3. Truss Support Loads

Support structure Element	loads	Maximum value	Deployed length	Applied loads	
Pitch loads				Mp (N·m)	Vp (N)
Pitch strut	Axial	430N	50 bays/100m	1671	18
Deployment Rail	Axial	260N	50 bays/100m	1671	18
	shear	405N	49 bays/98m	1620	17
	moment	364N-m	34 bays/68m	878	13
Roll loads				MR (N·m)	VR (N)
Upper strut	Axial	1065N	30 bays/60m	626	11
Lower strut	Axial	645N	50 bays/100m	943	10
Deployment	Axial	717N	50 bays/100m	943	10
Rail	shear	210N	49 bays/98m	931	10
	moment	121 N-m	44 bays/88m	820	10

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Truss loads for the revised truss configuration with a 250 kg tip mass and VRCS control moments applied by the Orbiter were determined to be very low, as seen in Figure 3-15. The slender struts used in the structure were determined to be compatible with the maximum loads indicated. The truss struts are manufactured from either a GY70/934 graphite epoxy material or a Pitch 75 type fabric to provide the high modulus in the laminated tubes ($E = 20 \times 10^6$ psi) required to minimize wall thicknesses for reduced cost and weight. The graphite epoxy material also provides the near-zero CTE required for thermal stability of the truss structure.

Maximum-pitch moment loads
(50-bays deployed)

M = 1.671 kN·m
V = 18 N

−566   −6   +6   +566N

+6   −6

−578   +578

418

+289   −289

2.0m

−289   +289

409

2.83m

Maximum-roll moment loads
(50 bays deployed)

M = .943 kN·m
V = 10 N

−226   −5   +5   +226N

236   236

−231   +231

−231   +231

+231   −231

231   231

2.0m
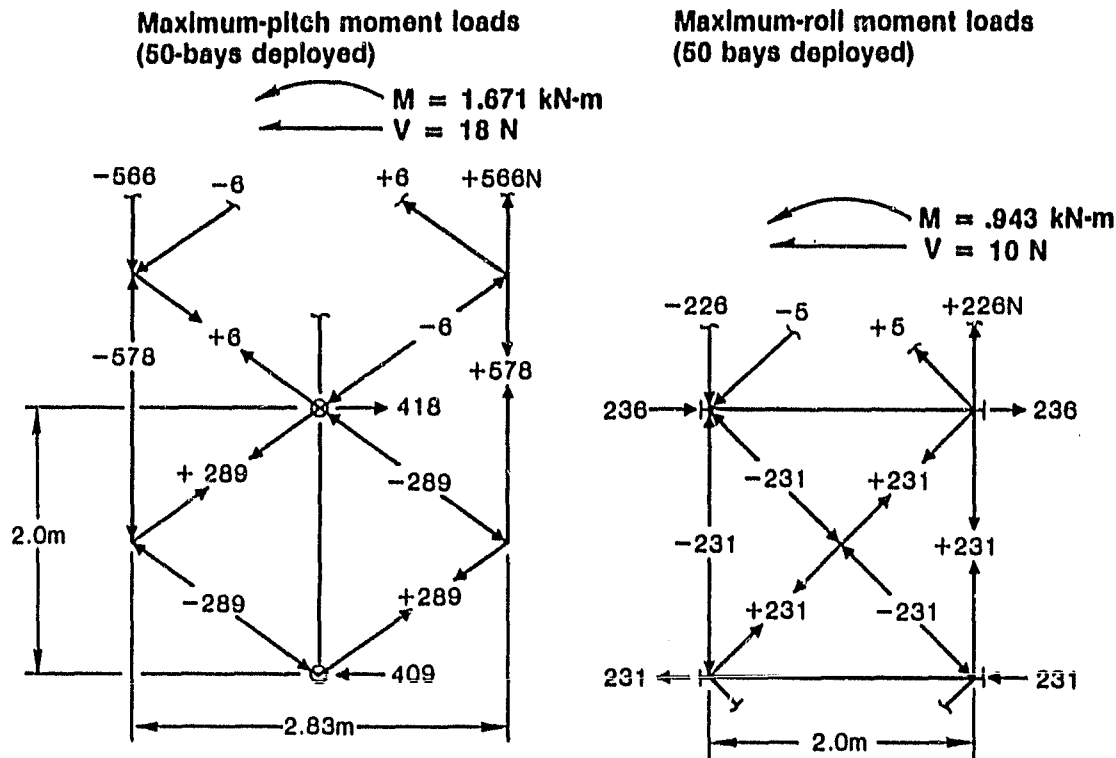
Figure 3-15. Revised Truss Loads

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3.3.2 **MASS PROPERTIES**. Mass properties for the revised experiment were calculated as shown in Figure 3-16. The moments of inertia are given relative to the Orbiter coordinates. The mass properties of the Orbiter are not included in these tables.

The c.g. change for the fully deployed truss with the tip masses ejected is shown for reference. With the tip masses ejected, the remaining dampers and support still constitute a tip mass of approximately 55 kg. Ejection of the complete tip package would lower the Z axis c.g. coordinate to 22.2 m.

3.4 SAFETY ANALYSIS

A preliminary phase 0 safety analysis of the SCE was conducted to identify the potential hazards based on the preliminary design data. This analysis forms the basis for identifying safety critical requirements for the experiment final design phase and assessing the adequacy of the preliminary design in conforming to Shuttle payload safety requirements. The Payload Safety Matrix, Hazard Lists and Payload Hazard Reports are included in this report as Appendix A.

The preliminary SCE hazards analysis of mechanical subsystems is summarized in Table 3-4. The two failure tolerant (2 F/T) functions that have been identified are basically compatible with the controls subsystem concept; however, detail mechanical functions will require further scrutiny during the final design to ensure 2 F/T capability.

GDC-ASP-82-004

Boost

Truss

Cradle & experiments

Operation

Item	Weight (kg)
Tip-mass	250
Truss	177
Cradle	657
Experiments	91
Total	1,175

Deployed phase	Center of mass (m)			Moment of inertia (kgm^2)		
	X	Y	Z	I_{xx} (Roll)	I_{yy} (Pitch)	I_{zz} (Yaw)
1/3	19.96	0	17.17	10.74×10^4	10.78×10^4	1.83×10^3
2/3	19.96	0	26.97	6.36×10^5	6.37×10^5	1.83×10^3
Full	19.96	0	38.74	1.59×10^6	1.59×10^6	1.83×10^3
Jettison tip-mass	19.42	0	27.2	7.64×10^5	7.65×10^5	1.2×10^3

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Figure 3-16. Revised Mass Properties

Table 3-4. SCE Mechanical Subsystems Hazards Analysis Summary

Subsystem	Hazard group	Hazard title	Hazard controls
Mechanical	Collison	Premature beam extension	• 2 F/T deployment control
		Premature jettison	• Jettison latches shuttle-provided
		Premature release of rail latches	• Structural safety factors • 2 F/T satch mechanism
		Premature release of tip mass	• Structural safety factors • 2F/T release control
		Orbiter cargo bay doors close prematurely	• Shuttle-provided
		RMS collides with STS	• Shuttle-provided
	Injury & Illness	Inadvertent retraction during EVA	• 2 F/T retention control
		RMS injures personnel during EVA	• Shuttle provided
		Damage to space suit during EVA	• Smooth all surfaces & edges • 2 F/T inhibit of mechanical motions
	Loss of entry capability	Payload blocks closure of cargo bay doors	• 2 F/T retraction control • 2 F/T jettison control

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The preliminary hazards analyses of electrical, material, structures and ground support equipment are summarized in Table 3-5.

The criticality of the structure to the safety of the Orbiter points to the need for very high standards of quality and materials controls. These items will have substantial cost impact on the flight structure. However, they are also necessary to achieve the modeling accuracies required for large space structures.

Table 3-5. Other SCE Subsystems Hazards Analysis Summary

Subsystem	Hazard Group	Hazard Title	Hazard Controls
Electrical	Electrical Shock	Personnel contact with electricity	● 28 Vdc power only
	Explosion	Rupturing of electronic packages	● Vent all packages
	Fire	Ignition of electronic packages and/or surrounding materials	● Current limiting ● Explosive atmosphere test
	Temperature extremes	Hot surface induced by excessive current flow	● Current limiting ● Power inhibited for EVA control
Material	Contamination	Offgassing of hazardous materials	● No equipment in AFD ● Eliminate ignition sources
	Fire	Flammable materials support combustion	● Use nonflammable/self-extinguishing materials
Structures	Collision	Failure of beam structure Failure of support structure	● RMS shuttle provided ● Beam is 1 F/T ● Materials control ● Corrosion control ● Structural safety factors
Ground Support Equipment	Collision	Loss of control during ground handling Structural failure of GSE	● Analyze loading procedures ● Loading equipment shuttle provided ● Materials control ● Corrosion control ● Structural safety factors

SECTION 4

DYNAMICS, CONTROL AND INSTRUMENTATION ANALYSIS

The initial requirements for Part I of this study resulted in a preliminary design of the experimental structure that incorporated high bending strength to accommodate potential failure modes in the Orbiter Reaction Control System. Cost considerations .nade it necessary to assume that nominal precision in the structural joints and minimum instrumentation approaches would be used. The resulting structural stiffness precluded meaningful Orbiter flight control interaction experiments and the instrumentation system did not address the issues of parameter identification. This section presents the results of analyses and investigation performed in response to new structural dynamic, control (both flight control and structural control), and identification system requirements generated for Part II of this study.

4.1 DIGITAL AUTOPILOT/STRUCTURE INTERACTIONS

During Part I of the study, the Orbiter Digital Autopilot (DAP) rate estimator was found to have frequency sensitive characteristics which largely determine whether or not oscillatory motions of the experimental structure will couple into the DAP. The DAP simulations are run at The Charles Stark Draper Laboratory (CSDL) using structural dynamics data supplied by Convair. Using data supplied by CSDL, the rate estimator was found to correspond to a second order filter with a natural frequency of 0.04 Hertz and a 0.8 damping ratio.

4.1.1 <u>CSDL SIMULATION RESULTS FOR PART I</u>. Late in Part I, data for a 50m structure with a flexible mount was developed and transmitted to CSDL, but the simulation results were not available in time to be included in the Part I Final Report. The flexibly-mounted structure was 50 meters long, had a 400 kilogram tip mass, and a mounting spring constant of 1.0×10^5 N-m/rad. This gave a first pitch bending frequency of 0.046 Hz and a first roll bending frequency of 0.07 Hz for the free-free structure-Orbiter combination.

A time history from the CSDL simulation is shown in Figure 4-1. At the start of the run a 10-degree roll maneuver at 0.2 deg/sec is commanded with the phase plane rate limit set at 0.02 deg/sec and the attitude deadband at 1.0 degree. (The traces are for the flexible body only, before the rigid body response is added.) Vernier Reaction Control System (VRCS) activity is indicated by the high frequency on the Z translation trace at the top. After 60 seconds, the rate limit is reduced to 0.01 deg/sec and the deadband to 0.1 degree. The traces show that, during the first 60 seconds, there is some VRCS action but it appear to have died out at the 60-second point. There are some modal oscillations, but these are decaying in amplitude.
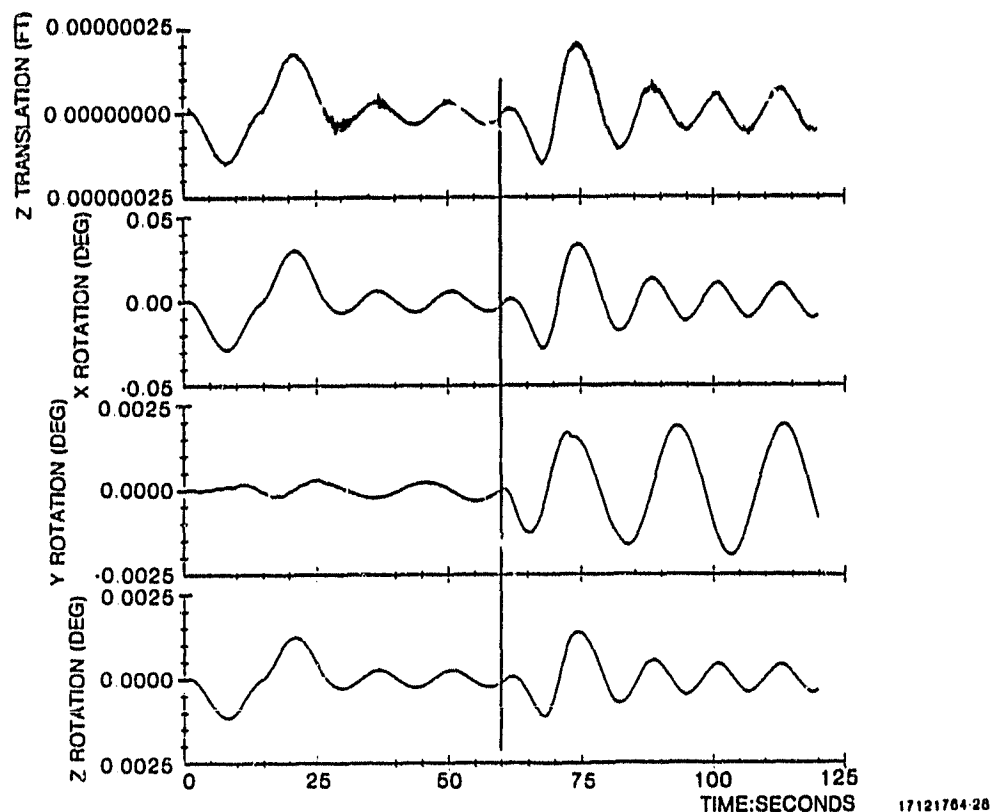
Figure 4-1. Interaction of Flexible Structure and the DAP

When the phase plane parameters are tightened at 60 seconds, the VRCS firings are seen to persist and the small amplitude Y rotation (pitch) does not seem to be damping out. Since the run is too short to fully characterize the pitch behavior, additional investigations are required.

There is absolutely no intent to operate close to any DAP instability but rather to achieve sufficient off-nominal operation to permit an evaluation of the structural modeling and DAP simulation as they apply to Orbiter-attached large space structure.

Subject to a more detailed pitch axis evaluation, the characteristics shown in Figure 4-1 appear to be very desirable. Normal operations can be carried out with the initial phase plane limits and the DAP behavior will be essentially nominal, but tightening the limits challenges the DAP and provides off-nominal behavior for the structural interaction evaluation.

4.2 EXPERIMENT MODAL EXCITATION

The SCE presents a new challenge in modal excitation in that the frequencies are quite low when compared to past structures. Although the experimental structure configuration has changed from time-to-time as the design require-ments have changed, the first bending modes have consistently been a fraction of a Hertz.

Vibration mode testing has commonly used electro-mechanical linear shakers for excitation, but it is generally conceded that these devices are not useful below 2 or 3 Hertz because of stroke and/or mass practical limitations. Nevertheless, the need to obtain data for evaluation of structural modeling techniques and for evaluation of system identification techniques requires that the low frequency modes be excited.

4.2.1 <u>EVALUATION OF ALTERNATIVE EXCITATION TECHNIQUES</u>. Three candidate excitation techniques were chosen for evaluation: Orbiter reaction control system (RCS) firings, a mass expulsion thruster system at the tip of the structure, and torque wheels. A 100m structure with a 1000 kg tip mass was selected for this analysis, based on availability of suitable dynamic data. The Orbiter-attached structure was then evaluated for the relative response of the first three pitch free-free bending modes. These first three modes had frequencies of 0.072, 0.92, and 3.0 Hz, respectively. Since the relative modal response is dependent on the type of measurement to be made, acceleration, velocity, and displacement were considered for both linear measurements (mode shape) and angular measurements (slope). The results are presented in Tables 4-1, 4-2, and 4-3.

Table 4-1. Relative Modal Excitation from RCS Firing

	Normalized gain		
	Mode 1	Mode 2	Mode 3
$\ddot{X}$ tip	1.00	.021	0.0040
$\dot{X}$ tip	1.00	0.0016	0.0001
X tip	1.00	0.0001	2×10^{-6}
$\ddot{X}$ max	1.00	0.431	0.135
$\dot{X}$ max	1.00	0.033	0.0033
X max	1.00	0.0026	0.0001
$\ddot{\theta}$ tip	1.00	0.810	0.436
$\dot{\theta}$ tip	1.00	0.063	0.011
θ tip	1.00	0.0049	0.0003

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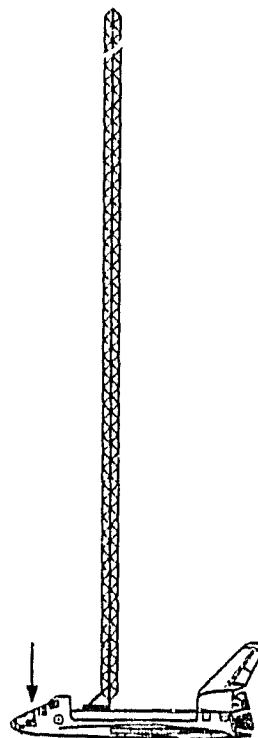
Table 4-2. Relative Modal Excitation from Tip-Mounted Thruster

PITCH MODES ONLY

	Normalized gain		
	Mode 1	Mode 2	Mode 3
$\ddot{X}$ tip	1.00	0.064	0.018
$\dot{X}$ tip	1.00	0.0042	0.0004
X tip	1.00	0.0003	10^{-5}
$\ddot{X}$ max	1.00	1.126	0.654
$\dot{X}$ max	1.00	0.087	0.016
X max	1.00	0.0068	0.0004
$\ddot{\theta}$ tip	1.00	2.12	2.12
$\dot{\theta}$ tip	1.00	0.164	0.051
θ tip	1.00	0.013	0.0012

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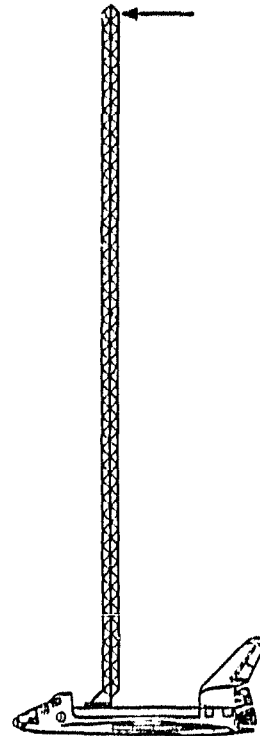
Table 4-3. Relative Modal Excitation from Tip-Mounted Torque Wheel

PITCH MODES ONLY

	Normalized gain		
	Mode 1	Mode 2	Mode 3
$\ddot{X}$ tip	1.00	2.12	2.12
$\dot{X}$ tip	1.00	0.164	0.051
X tip	1.00	0.013	0.0012
$\ddot{X}$ max	1.00	43.68	75.68
$\dot{X}$ max	1.00	3.39	1.83
X max	1.00	0.26	0.044
$\ddot{\theta}$ tip	1.00	82.1	245.2
$\dot{\theta}$ tip	1.00	6.40	5.94
θ tip	1.00	0.49	0.144

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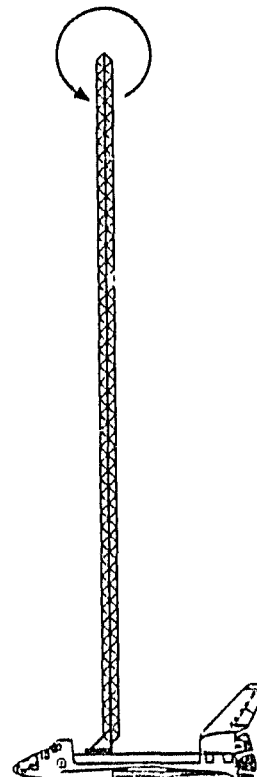
All values have been normalized to the first mode. For example, in Table 4-1 the maximum displacement (X max) of the third mode is seen to be 2×10^{-6} times the maximum displacement of the first mode. Thus, it appears that attempts to gather data from the higher modes by firing the RCS and taking displacement measurements (as might be taken by an optical system) can be expected to present problems in extracting the third mode signal from the first mode noise.

4.2.2 SELECTION OF EXCITATION TECHNIQUES. Inspection of the three tables indicates that the RCS tends to excite mostly the first mode, that the thruster at the tip is somewhat better than the RCS for higher mode excitation, and that a torque wheel at the tip is by far the best technique for providing reasonably uniform modal excitation. Based on these results, the RCS was chosen as the SCE technique to excite the structure for the free decay tests wherein amplitude-sensitive nonlinear behavior (damping) will be observed, and torque wheel excitation was chosen for the random shake to produce multi-mode data.

Based on the results of the preceding analysis and on knowledge of available instruments and their applications, linear acceleration and angular rate were selected as the parameters to be monitored.

4.2.3 MODAL EXCITATION WITH RCS. Although normalized data is most easily evaluated for relative effects, absolute data is needed to determine if reasonable amplitudes are achieved. Analysis of step firings of VRCS thrusters R5D and L5D produced a first pitch bending maximum displacement of 0.083 meter and a maximum second mode displacement of 0.0002 meter. Thrusters R5D and L5D point down from the aft end of the Orbiter and produce negative pitch accelerations. By terminating the thrust after half a period of first mode oscillation, the first mode displacement would double to 0.16 meter. If additional amplitude were desired, the process could be repeated using F5R and F5L (forward down pointing for positive pitch accelerations) and the amplitude would again approximately double to 0.33 meter.

The effect of firing the Primary RCS (PRCS) is also of interest, provided excessive structural loads are not induced. Using an effective thruster-on time of 0.020 second to determine the minimum impulse bit (MIB), it was found that simultaneous MIBs from PRCS thrusters F1U, F2U, F4D, and L4D produced a tip displacement of 0.25 meter (thrusters F1U and F2U fire upward from the front end and F4D and L4D fire downward from the aft end). All four thrusters produce negative pitch acceleration. It is tentatively concluded that very limited PRCS MIB operations can be carried out without producing excessive experiment structural bending, pending CSDL loads evaluation.

4.2.4 MODAL EXCITATION WITH TORQUE WHEELS. Actuation or "muscle" for control and excitation in space at low frequencies can be provided by torque wheels. Figure 4-2 shows the relations for a simple dc motor where

L   is the inductance in henries

R   is the resistance in ohms

K_T is the torque constant in the ft·lb per amp

K_B is the back of emf constant in volts/rad/sec

J   is the rotor moment of inertia in slug-ft-sq

s   is the Laplace operator

The wheel can saturate speed-wise when the back emf cancels the applied voltage such that current (and torque) drop to zero. This is exhibited in the torque/voltage frequency by a low frequency roll-off. Using the block diagram of Figure 4-2 to solve for the break-point or corner of this roll-off, it can be shown that the corner frequency occurs at $K_B K_T/(JR)$ rad/sec. Thus, by increasing the rotor moment of inertia, J, the low frequency performance can be achieved. Using motor parameters of $K_B = 0.11$, $K_T = 0.114$, and $R = 1.6$, a 0.048 Hz corner can be achieved with a J of 0.027 slug-ft-sq. This corresponds to an 8.0 inch diameter rotor weighing 16 pounds.
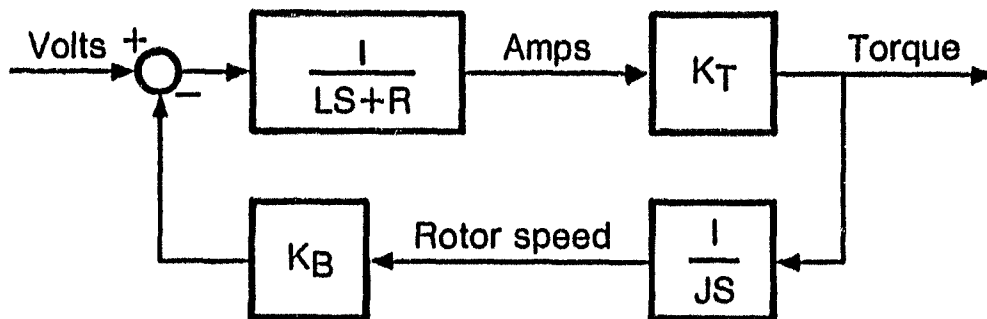
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Figure 4-2. Torque Wheel Block Diagram

4.2.5 JOINT CLEARANCE EFFECTS. The effects of joint clearances in producing measurable backlash (accumulated clearances) were discussed in Part I of the study (reference 1, subsection 3.2.19). The Part II a zero backlash structure is required to ensure the tip position accuracy needed for large space atnenna feed masts.

In the technology of large deployable structures the ability to control and predict the joint clearance effects on damping and nonlinear vibration behavior of structures in space is essential. It is, therefore, desirable to conduct an experiment with the SCE that will validate this technology. This could involve inducing clearances in a sufficient number of joints at the conclusion of all other structural dynamic and control tests.

A concept for unloading the interference fits in a number of the test truss
joints is shown in Figure 4-3. The eccentric pins in an expandable bushing
would maintain zero clearance fit in each test joints until the pins are rotated
either by remotely activated cables or by manual EVA action. The truss would
initially be excited in the first pitch bending mode by firing the RCS thrusters.
The free decay characteristics of the structure would be measured for a sufficient
period of time to establish a damping ratio. After unloading the test joints this
test sequence would be repeated to obtain comparative results.

Joint clearance effects
- Accumulative truss backlash
- Contributes to damping
    - Sliding friction
    - Impact energy
    - Air compression

Joint unloading concept

Eccentric pin

Expandable bushing

Rotate to increase joint clearance } Manual or rigged operation

Install in 3-4 clusters (18-24 joints)
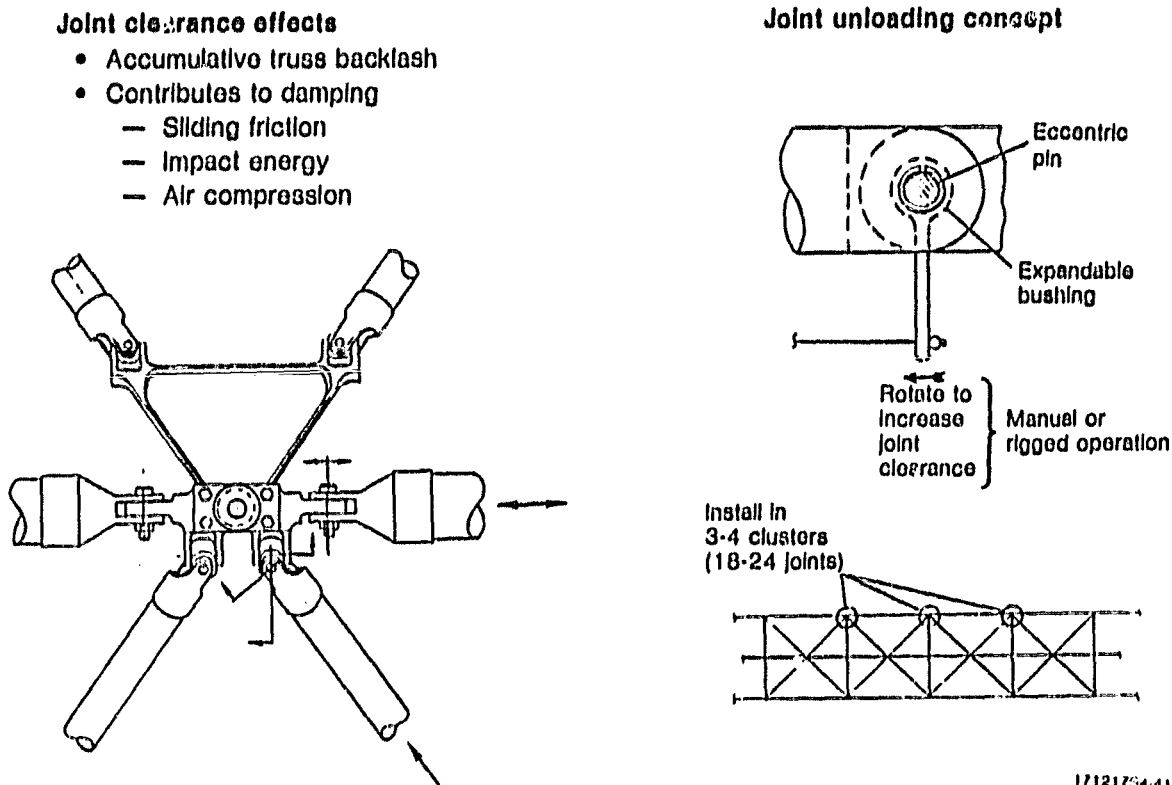
Figure 4-3. Joint Clearance Effects Test Concept

4.2.6 CLOSELY SPACED COMPLEX MODES EFFECTS. Since a mast will have a
sparsely populated modal spectrum, a "feed array" platform concept was designed
to provide the modal density which is typical of antenna reflectors. Based on
experience with the Convair experimental control platform, a simple platform can
be designed with two closely-spaced modes at about 0.2 Hz and three closely-
spaced modes near 0.7 Hz. Thus, the frequencies of a large reflector could be
matched but the masses and mode shapes would be different.

The simple platform would be attached to the tip of the mast as shown in Fig-
ure 4-4. The platform would fold for stowage and be deployed by the RMS.
Four torque wheel damper sets would be installed on the platform to permit
multi-modal excitation for test and post excitation modal damping.
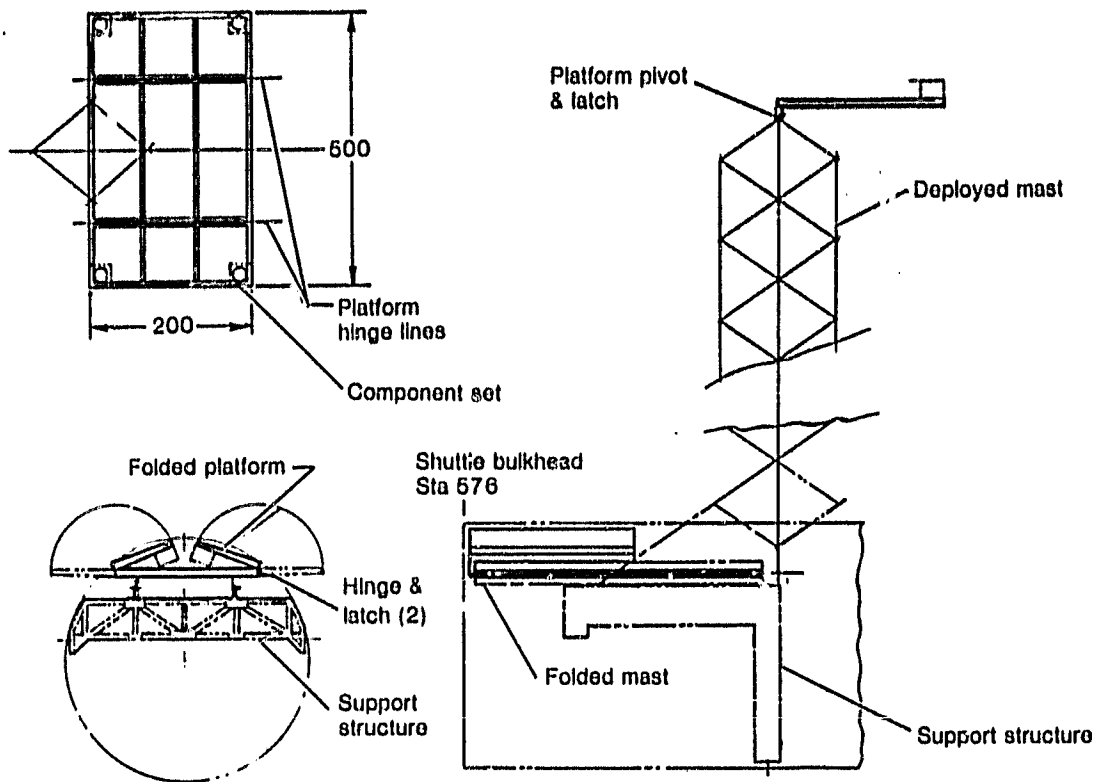
Figure 4-4. Feed Array Platform Concept   17121704-42

4.2.7 SUMMARY OF CONTROL AND DYNAMICS EXPERIMENT OPTIONS. The approaches considered thus far provide the capability to address a number of structural dynamic and control issues. In Table 4-4, checks indicate areas of investigation, circles indicate a selected approach. A yaw maneuver was not selected since there appears to be no problem in that axis (this should be confirmed by future simulations). Sinusoidal torque wheel excitation has not been selected since searching out a single mode can be extremely time consuming. Investigation of closely-spaced complex modes with the "feed array" platform structure has not been selected since closely-spaced mode issues can be addressed on the ground.

4.3 INSTRUMENTATION

The SCE will be instrumented to measure the parameters necessary to identify and accurately quantify mode shapes and modal frequency response of the first six modes of the test truss attached to the Orbiter in space free flight. This will require measuring linear and angular displacements and rates at selected stations along the structure as well as Orbiter motions and the relative motion between the structure and the Orbiter interfaces. The linear displacement of the tip of the test truss relative to the base of the truss will also be measured to verify the precision with which relative tip motion can be maintained.

Table 4-4. SCE Control and Dynamics Options

Excitation/Test Method	Structural Dynamic and Control Issues				
	Linear Dynamics	Nonlinear Behavior	DAP Interactions	Control Device Demo	Identification and Control of Complex Modes
Orbiter Maneuvers (RCS)					
Pitch	✔	(✔)	(✔)	(✔)	
Roll	✔	✔	(✔)	(✔)	
Yaw	✔	✔	✔	✔	
Torque Wheel Random Shake	(✔)			(✔)	
Sinusoidal Torque Wheel	✔	✔		✔	
RCS on Mast Tip	✔	✔			
"Feed Array" Structure	✔			✔	✔
Variable Joint Clearance		(✔)			

(✔) Selected Options.

4.3.1 MODE SHAPE INSTRUMENTATION. Mode shape instrumentation is required for evaluations of structural modeling accuracy and for the generation of system identification data.

Since acceleration is related to mode shape (displacement) by frequency squared, the low frequency modes of the fully deployed truss structure will have small accelerations and require precision servo accelerometers. The partially deployed test configuration will have higher frequencies so the frequency squared effect will permit use of less costly accelerometers such as piezoelectric.

The first step in establishing mode shape instrumentation is to define the quantity and installation location of the servo (force balance) accelerometers. The placement of piezoelectric accelerometers must await further definition and analysis of the partially deployed structure. The servo accelerometers are required to measure the small accelerations of the first modes: they are more sensitive and more expensive than the piezoelectric units. The repeatability of the servo devices is commonly expressed in micro g's whereas piezoelectric repeatability is in the milli g range.

The mode shapes for the first three pitch bending modes of the structure are shown in Figure 4-5. Although the Orbiter is quite massive when compared to the structure, it does respond enough to change the first mode from the classical cantilever shape. The instrument placement shown covers all maximums and provides two measurements at all nodes (zero displacement) except the very shallow node of the first mode. Two measurements near a node permit interpolation to more accurately locate the node point.

0.086 Hz

1.18 Hz

Base

Tip

3.72 Hz

Accelerometer locations ⟶

Location meters	30	54	58	70	98	100
1st pitch	6.2	30.1	35.2	64.3	96.7	100
2nd pitch	59.0	99.5	100	70.2	0.3	−7.7
3rd pitch	100	7.6	−18.7	−92.8	−9.3	4.4

Percent of maximum displacement
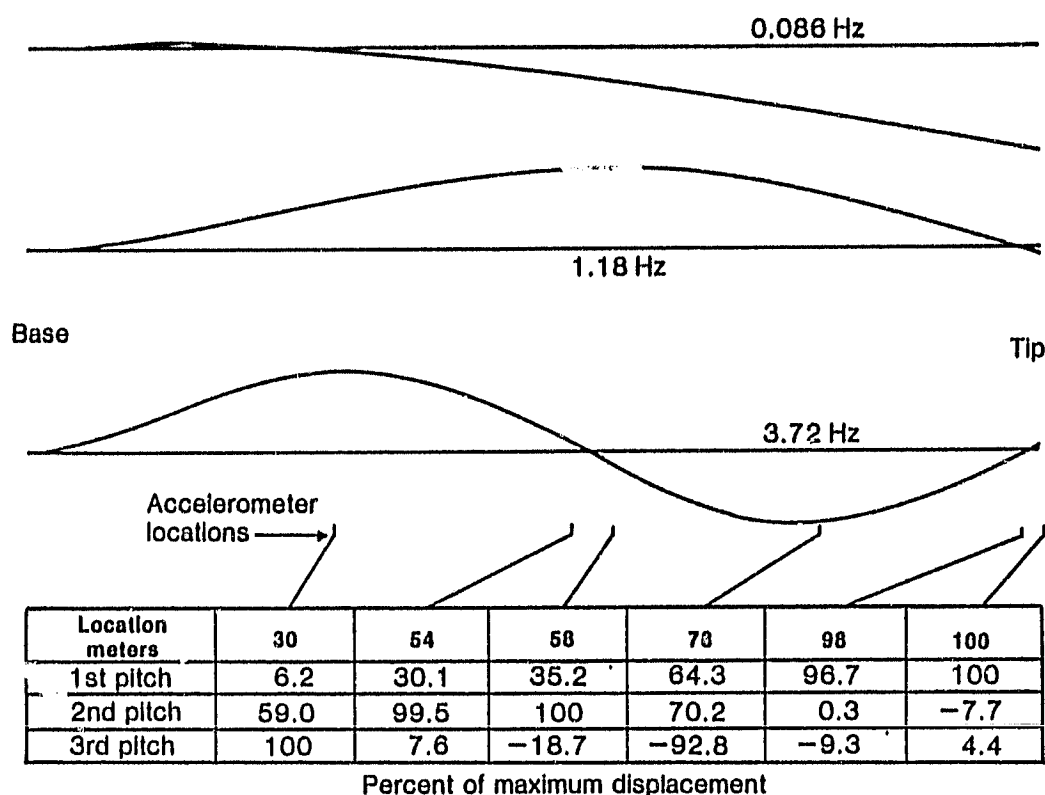
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Figure 4-5. Pitch Bending Mode Shape Instrumentation

Figure 4-6 shows the roll bending modes. Since the Orbiter roll moment of inertia is much smaller than the pitch inertia, the first roll bending departure from classical cantilever is more extreme than for the pitch axis. The second and third modes in both roll and pitch are very close to cantilever behavior. Using the same stations along the structure for the roll axis instruments as were used in pitch, all maximum displacements are measured and all nodes have two measurements.

In addition to accelerometers to measure the mode shape (translation), rate gyros are needed to define the modal slope (angle) at the excitation input point. Since torque at the tip of the structure has been selected for excitation, slope at the tip is required to determine the modal response in generalized coordinates which is used, in turn, to calculate the mode shape coefficients. The torque wheels will have colocated rate gyros for active damping feedback. These same rate gyros will be used for slope instrumentation since they are already in the proper location.

Three additional rate gyros, one in each axis, are installed at about 78 meters above the base to provide additional slope data for the low frequency modes where the accelerations will be small as compared to the higher frequency modes.

0.114 Hz

0.935 Hz

Base   Tip

2.89 Hz

Accelerometer
locations ———▶

Location meters	30	54	58	78	98	100
1st roll	−31.7	−8.3	−1.3	42.2	94.6	100
2nd roll	61.1	99.8	100	69.6	0.1	−7.7
3rd roll	100	1.2	−2.4	−92.5	−8.9	4.5

Percent of maximum displacement
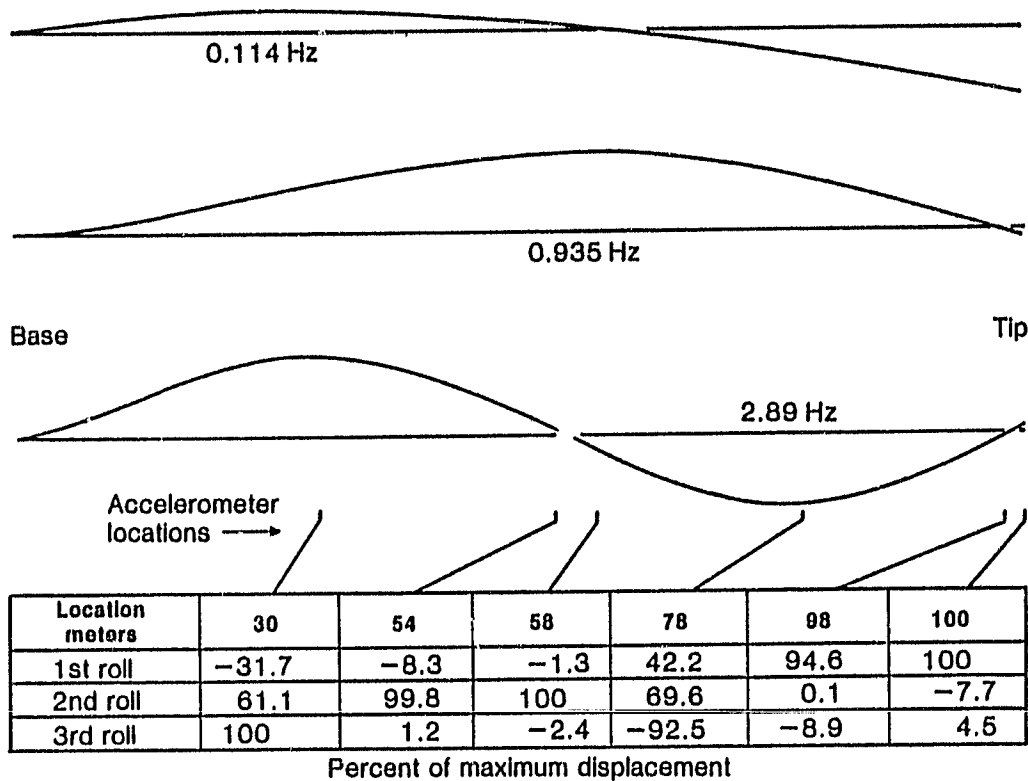
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Figure 4-6. Roll Bending Mode Shape Instrumentation

4.3.2 TIP MOTION INSTRUMENTATION. Large antenna feed mast technology requires that the tip of the structure be within 10 cm of its nominal position. Concepts for determining the location of the tip are shown in Figure 4-7 and are described as follows:

a. Transverse and torsional displacement measurements.

Concept 1, CCD Camera System. A long focal length lens is attached to a CCD camera and mounted at the base of the structure. This camera views a passive target mounted at the far end of the beam. The position of the image of the target on the CCD array can be determined using simple image processing techniques and this information can be used to calculate the position of the beam tip in a plane normal to the optical axis of the camera. Using a 1000 mm focal length lens, the CCD array will view a rectangular area approximately 1m × 0.85m at the end of the beam. The position of the target can be found within 0.5 cm using current CCD arrays. The design of this system will involve studies to find techniques for minimizing camera tilt effects, trade studies on the merits of intensified CCD cameras and the development of image processing algorithms to calculate the target position.

Concept 1 — Transverse & Torsional Displacement
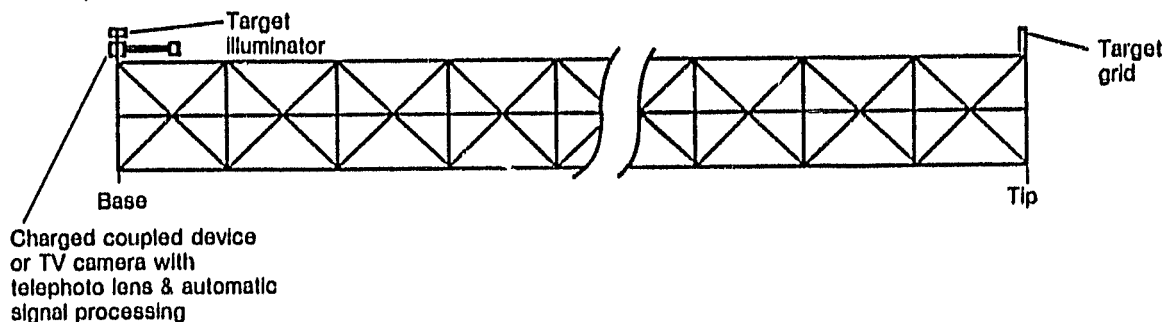
Concept 2 — Transverse & Torsional Displacement
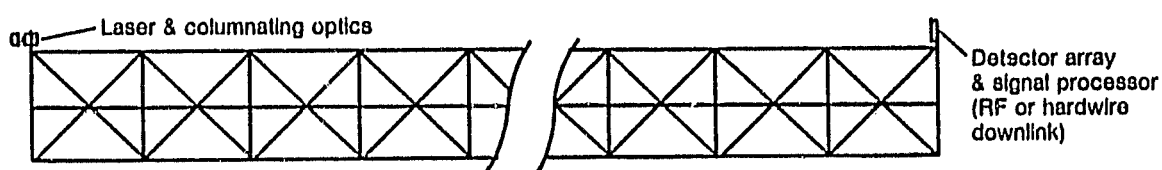
Concept 3 — Longitudinal Displacement
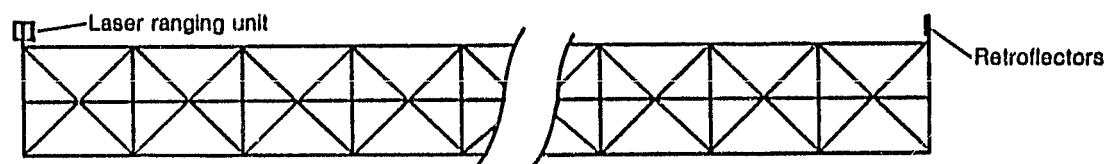
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Figure 4-7. Tip Motion Measurement Concepts

Concept 2, Laser/Detector Array System. A laser (probably a laser diode) and a beam expanding telescope are mounted at the base of the structure. The laser beam is directed along a fixed direction and it intercepts a detector array mounted at the far end of the beam. The array consists of a large number of independent silicon photo-diodes mounted on an area approximately one meter square. The signals from the detectors are decoded by the signal processing electronics to determine the position of the beam on the array.

A modification of this technique is to make a small array and put some type of beam steering mechanism on the laser source. The laser beam is moved until it is centered on the detector array and the beam tip position is calculated from the pointing direction of the laser.

b. Longitudinal displacement.

Concept 3, Conventional Laser Ranging System is mounted at the base of the structure. These instruments are available from several companies and they are presently used for surveying. The target for the ranging system is a number of retroreflectors mounted at the tip of the beam. The range to the target can be measured within 0.3 mm. The only required modification of the laser ranging system is to make it space qualified.

c. Transverse torsional and longitudinal displacement. These measurements could be easily combined by mounting a steering device on the laser ranging system. The pointing direction of the sensor when it sees the target can be used to calculate the transverse position and the normal ranging output provides the longitudinal position.

4.3.3 MEASUREMENT OF ORBITER MOTIONS. The planned on-orbit capability to measure Orbiter motions for the SCE flight time frame consists of a calculated state-vector updated at approximately 6.25 Hz. This capability is required to support advertised payloads navigation update, but the sample rate will probably not provide adequate resolution for the SCE tests.

The raw data from the Orbiter rate gyros will be available in downlist. This will require an off-line ground support system to format the data into pitch, yaw and roll rates.

4.3.4 FORCE AND MOTION MEASUREMENTS AT ORBITER INTERFACE. Force and rotation measurements at the Shuttle Orbiter/SCE interfaces may be measured directly at the trunnion pins provided for the standard five point retention system of the SCE shown in Figure 4-8. Pin loads would be measured by strain gauges attached to each pin while pin motions would be measured with linear potentiometers, provided such motions prove to be significant.

The roll braces and pitch braces would also be instrumented with strain gauges or load cells. This loads data would allow the deflections of the support structure to be computed from its structural model. However, static ground tests of the support structure are recommended to validate its model.
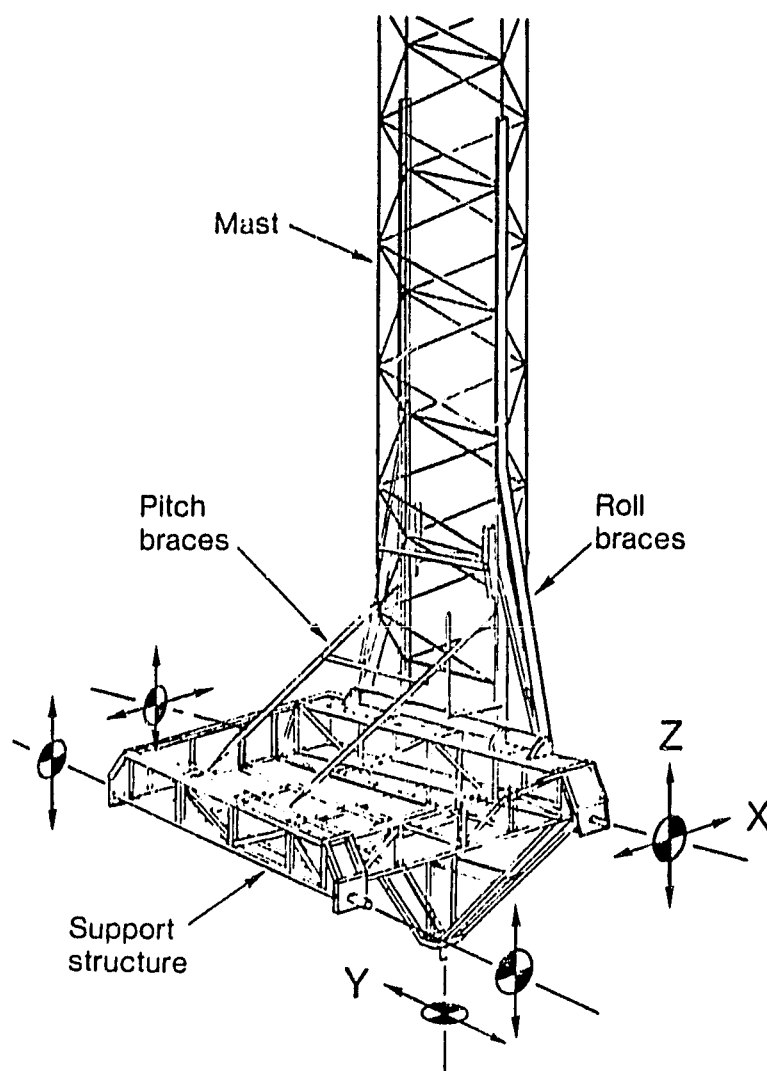
Figure 4-8. Forces at the Orbiter Interfaces

SECTION 5

PRELIMINARY SYSTEM TEST PLAN

5.1 INTRODUCTION

The Space Construction Experiment (SCE) is a basic early Shuttle flight experiment that will develop and test the capabilities of the Space Shuttle system to support construction of large space systems. The basic SCE will consist primarily of a large deployable structure equipped with controls, instrumentation, and representative subsystems elements to allow testing of Orbiter control during and after construction, construction operations using basic Orbiter capabilities, and predicted dynamic behavior and control of a large deployable structure attached to the Orbiter.

The SCE will be integrated into the Shuttle as a secondary payload of opportunity. Flight testing is to be performed on a non-interference basis with primary payloads.

5.1.1 PURPOSE.
The purpose of the System Test Plan (STP) is to provide the policies, plans, and overall requirements for the testing to be accomplished for the SCE program. The STP encompasses all levels of testing to be performed in the SCE program. This includes development testing, qualification testing, acceptance testing, flight certification testing, ground operation testing, and flight test operations and instrumentation.

5.1.2 GROUND RULES AND ASSUMPTIONS.
The SCE test program shall be conducted in accordance with the following round rules and assumptions:

a. Only one SCE test article will be produced for ground and flight testing.

b. Major ground simulation tests are planned using LaRC and JSC facilities.

c. The SCE support structure interface test will be conducted by LaRC.

d. Flight certification testing will be primarily performed at the system level to minimize the cost of verifying overall flight worthiness of the experiment.

e. Major flight certification tests are planned using JSC facilities.

f. The flight test operations will be conducted aboard the STS Space Shuttle Orbiter.

5.1.3 <u>TEST PROGRAM SUMMARY</u>. The test program flow diagram (Figure 5-1) describes an orderly progression to meet the SCE program objectives and requirements. This test program is required to assure the performance of the flight experiment hardware and to verify the technologies required to accurately predict flight test performance of the structure, structural damping subsystem, Orbiter flight controls and manual construction and repair operations in space.
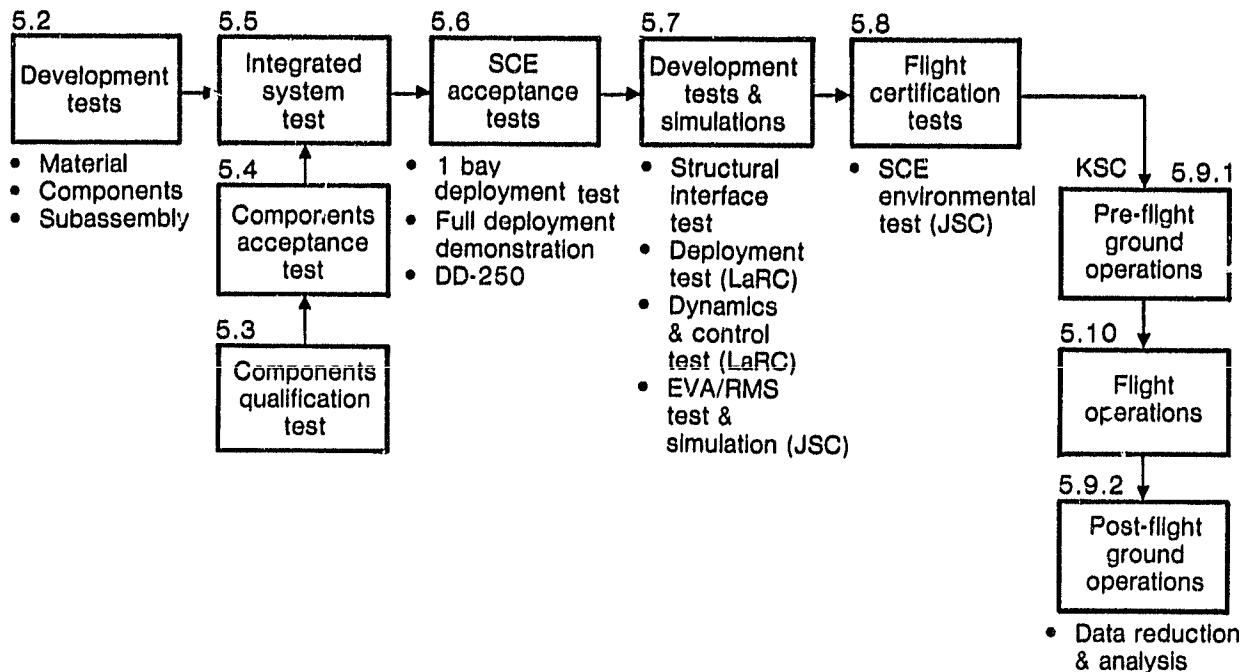
Figure 5-1. SCE Test Program Flow Diagram

The development testing phase will allow system manufacturing and design problems, and math modeling uncertainties, to be evaluated and resolved during the design phase. The component qualification testing will verify that no critical weaknesses exist before subsystem and system level tests are initiated. The flight certification tests will verify the flight worthiness, environmental compatability, and functional capability of the integrated SCE.

5.2 DEVELOPMENT TESTS

Development testing for SCE is planned to provide early solution to manufacturing and design problems, and to identify key characteristics of hardware. Materials, components, and subassemblies will be tested in progressive stages to ensure earliest recognition of possible problem areas.

Adapting existing flight-qualified torque wheels and rate gyros to this application will be a long-lead-time consideration. Manufacturing of the deployable truss will be a major cost driver and will require some technology development to achieve a cost-effective precision design.

Structural tests utilizing a 5-bay truss segment will ensure compatibility of the final truss design with the operational environment. It will also allow structural dynamics characteristics to be measured for verification and refinement of the math model for full-scale assembly performance predictions.

5.2.1 MATERIALS TESTS. Truss tube and fitting composite material specimens will be tested to measure mechanical properties and outgassing characteristics. Preproduction tube specimens and node fitting material test coupons will be tested to establish longitudinal and transverse tensile strength, compression strength and modulus; shear strength and shear modulus; and coefficient of thermal expansion (CTE) characteristics over the full range of operating temperatures. Truss composite materials, adhesives, bonding agents and other non-metallic materials will be tested or otherwise verified to be in accordance with Space Shuttle payload requirements for toxicity, outgassing, and vacuum stability.

5.2.2 COMPONENT TESTS. The following component tests will be performed:

a. Component Tests to Support Structural Dynamic Modeling. Structural components to be tested are shown in Table 5-1. The basic information which is required to simulate each component consists of the axial spring rates of the struts and braces, the cross-sectional moments of inertia of the deployment rails, a stiffness or flexibility matrix for the joint fittings, and the weight of each of the components. With the exception of the moment -of-inertia, each of these characteristics can be measured statically. Measurements of the concentrated masses will include the mass moments-of-inertia about the three basic axes. Sufficient quantities of each strut and node fitting configuration will be tested to establish a statistical population of values.

Table 5-1. Structural Components to be Characterized

Item	Tests and Measurements
Struts	
Node Fittings	
Pitch Brace	Spring rates and
Roll Braces	mass properties
Tip Package	
Deployment rail moment of inertia	Free-free vibration response

Cross-sectional moments of inertia are not directly measurable quantities and, thus, they must be obtained indirectly. A comparatively easy method of obtaining these parameters is to support the beam on wires located at or near the nodal points of the first free-free mode and then shake the beam to excite this first mode. Using the first frequency thus obtained, the cross-sectional moment-of-inertia may then be calculated.

b. **Truss Strut and Node Fitting Assemblies Tests.** Preproduction samples of each truss strut configuration and node fitting configuration will be subjected to a series of tests as follows:

1) Joint coupling effects of each pin joint configuration will be performed to measure joint behavior under static and dynamic loading conditions in the expected environment of temperature cycling and vacuum. Zero free play, thermal conductivity and electrical conductivity across each joint and hinge will be evaluated. Node-to-node thermal stability will be measured for conformance to near-zero CTE requirements. Joint swiveling torques will be measured. Bonded joint integrity will be verified.

2) Buckling stability and post buckling strength of each strut configuration will be measured. Strut specimens will be tested to failure in tension and compression.

3) Node joint ultimate strength tests under representative loading conditions will be performed on samples of each node fitting configuration.

c. **Damper Set Tests.** An engineering test article of a torque wheel actuator assembly will be assembled using a space qualifiable torque motor and rate gyro and connected to a simple control system. Damping performance of low modal frequencies will be evaluated using a simple cantilevered beam.

5.2.3 **SUBASSEMBLY TESTS.** The prediction of the dynamic response of the SCE requires the development of a finite element simulation of the system. This digital model may then be used to predict the dynamic response of the system due to excitations such as the forces and moments generated by the vernier reaction control system. MSC/NASTRAN is the basic finite element system which will be used and is basically a structural simulation made up of elements such as bar, tubes, rods and concentrated and distributed masses. In order that confidence may be gained in the adequacy of this digital model to simulate the "real world," it is necessary that ground tests be accomplished which verify this simulation.

Upon completion of the component tests, the next step is a vibration test of a separately manufactured segment of the SCE truss. The test truss will be mounted vertically as a cantilever and excited with electrodynamic shakers over a frequency range of essentially zero to 25 Hz as shown in Figure 5-2. Natural frequencies and mode shapes will be obtained and compared with the eigenvalues and eigenvectors which will be obtained from a finite element analysis of the truss segment. Use will be made of the component tests in assembling this finite analysis simulation and the total procedure will be a step in gaining confidence in the ability to predict modal frequencies and mode shapes of the full flight article.
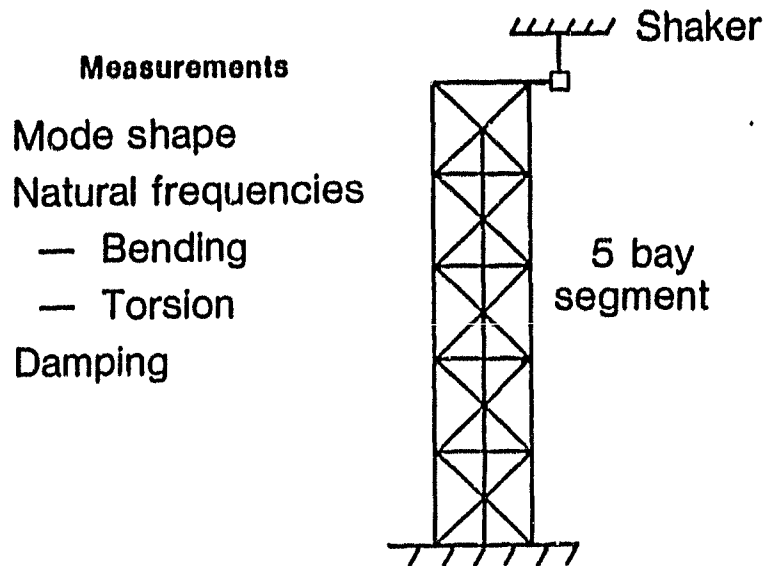
Figure 5-2. Truss Segment Test

At the conclusion of the dynamic testing a load fixture will be installed on the upper end of the truss segment. Static proof loads will be applied axially in each direction then torsionally in each direction. Static loads and truss tip deflections will be measured. Strut loads will be measured by attached strain gauges. This test will demonstrate the ability of the truss to withstand predicted flight structural loads, correlate axial and torsional stiffness results with that predicted by the structural model and evaluate strain gauge measurement techniques for strut loads.

5.3 COMPONENT QUALIFICATION TESTS

Component qualification testing is intended to assure the success of subsequent subsystem, system, and flight testing. All test specimens will have successfully completed a functional checkout and acceptance testing including burn-in (if required) before qualification testing.

Environmental qualification test requirements will comply with JSC-07700, Volume XIV (Revision G, September 26, 1980), "Space Shuttle System Payload Accommodations." All newly designed components will be qualified and existing qualified components will be reviewed and retested as required to ensure full compliance with Shuttle requirements. Components environmental testing will be minimized by performing major tests at the integrated system level during Flight Certification Testing to preclude numerous individual component and subassembly tests.

Qualification tests are sumarized in Table 5-2.

Table 5-2. Component Qualification Test Program Summary

	Ambient Operating	Vacuum or Thermal Vacuum	Vibration	Acoustic	EMC	Shock
Damper Package	X	X	X		X	
Deployment Carriage	X	X			X	
Control Unit	X	X	X	X	X	X
Tip Mass Ejector	X	X	X		X	X

5.3.1 DAMPER PACKAGE. The damper package, consisting of six torque wheel/rate gyro actuators will be functionally tested in ambient conditions to set up the phasing. The package will be tested for EMC and subjected to a functional thermal vacuum test and vibration test.

5.3.2 DEPLOYMENT CARRIAGE. The deployment carriage will be run through a long series of operating cycles in thermal vacuum to confirm its durability and reliability. It will also be tested for EMC.

5.3.3 CONTROL UNIT. The control unit and CU software will be functionally tested by supporting the damper and carriage tests. It will also be vibration tested, acoustic tested, EMC tested and shock tested.

5.3.4 TIP MASS EJECTOR. The tip mass ejector will be functionally tested in both ambient and thermal vacuum environments. The unit will be demonstrated in the vacuum environment after being subjected to vibration and shock testing. EMC testing will also be performed.

5.4 COMPONENT ACCEPTANCE TESTS

Component acceptance tests are formal tests required to demonstrate that the hardware and associated data is in compliance with specifications and ready for

delivery to NASA or for qualification test. These tests are designed to detect deficiencies in workmanship, material or quality. They are normally nondestructive in nature and performed on all deliverable units. They include functional testing and may include environmental testing if necessary to verify performance.

5.5 INTEGRATED SYSTEM TEST

System tests performed after final assembly and checkout of the integrated SCE and before final acceptance test, are described below.

5.5.1 AVIONICS SYSTEM INTEGRATION TEST.
Avionics system integration and evaluation test will be performed to demonstrate functional compatibility between the Control Unit, drive latch solenoids, carriage drive motors, caging drives, damper sets, sensors, PCM encoder, and all other Avionics data, power and control interfaces. This test will demonstrate and validate the CU software.

5.5.2 PARTIAL DEPLOYMENT/RETRACTION TEST.
Repeated partial deployment and retraction tests will be performed to evaluate the effects of deployment rates and accelerations on the behavior of the deployment drive mechanisms and the truss structure.

With a counterweight rigged to counteract the weight of the tip mass, the truss will be vertically deployed two full bays, then fully retracted. Drive rate profiles will be varied until an optimum performance is achieved.

5.6 SPACE CONSTRUCTION EXPERIMENT ACCEPTANCE TEST

Prior to acceptance and delivery of the Space Construction Experiment and associated end items, a series of formal acceptance tests will be conducted. These tests will be witnessed by the NASA and will culminate upon delivery of test data demonstrating performance of equipment to prescribed test specifications.

After final integrated system testing, the acceptance test will include, but not be limited to the following identified tests.

5.6.1 FULL DEPLOYMENT/RETRACTION TEST.
Tests of the deployable truss and deployment/retraction mechanism will be conducted in the horizontal position. Deployment and retraction will be with the aid of support dollies on low friction rollers. The truss will be fully deployed and fully retracted three times.

Electrical interface compatibility tests will be performed on the Power, Telemetry, and Data Services, and Displays and Controls interfaces. The commands to the Control Unit will be by the portable switch panel throughout the test. Monitoring of all applicable parameters will be provided by the contractor.

5.6.2 SUITCASE EXPERIMENT FIT CHECKS. With the truss fully deployed, prior to the final retraction test, the suitcase experiment hardware will be installed on the structure to demonstrate the fit-up and interface compatibility of the experimental hardware with the truss structure. The experimental hardware will be removed prior to final retraction.

5.7 GROUND TESTS AND SIMULATIONS PLAN

A simulation and ground test program plan which would fully develop modeling techniques for flight performance predictions would include the elements shown in Figure 5-3. The initial structural dynamics model will derive data on struts, joints, fittings, mass properties, etc., from the component tests. The model will be tested by performing subassembly tests of the modeled 5-bay structural segment. Structural interface tests of the flight experiment support structure will allow interface deflections at the base of the truss to be computed from measured flight loads. Deployment tests and dynamics and controls tests will allow the structural dynamic and control models for the flight test article to be evaluated and provide a data base for evaluating the effectiveness of ground test of partially deployed configurations in ensuring accurate flight test performance predictions.

5.7.1 DEPLOYMENT TEST. The deployment test will evaluate the effects of deployment rates and acceleration on the behavior of the structure and to finalize the functional operating parameters for the deployment/retraction mechanisms and controls in a simulated zero-g condition. The test will consist of varying drive rates and rate profiles and measuring the loads and disturbances in the truss structure. The test fixture as shown in Figure 5-4 will consist of a synchronized deployable suspension system. The suspension cables will be translated in unison with the truss structure by using a truss deployment carriage digitally controlled by the control unit.

5.7.2 STRUCTURAL DYNAMICS AND CONTROL TEST. The partially deployed structure (Figure 5-4) will be used to conduct a series of dynamic and controls tests. For a horizontal excitation it is necessary to ensure that the pendulum frequency of the zero-g suspension cables is well below the lowest modal frequency of the structure. This limits the length of structure that can be tested, unless very long suspension cables can be accommodated at the test facility.

The approach used for the structural dynamics ground test will recognize that the suspension will be part of the ground test dynamic system. Adjusting the model of the entire system to match test results should give the proper mass and stiffness matrices for the flight structure. The deployed structure will also provide an opportunity to check out active damper performance and component installation.
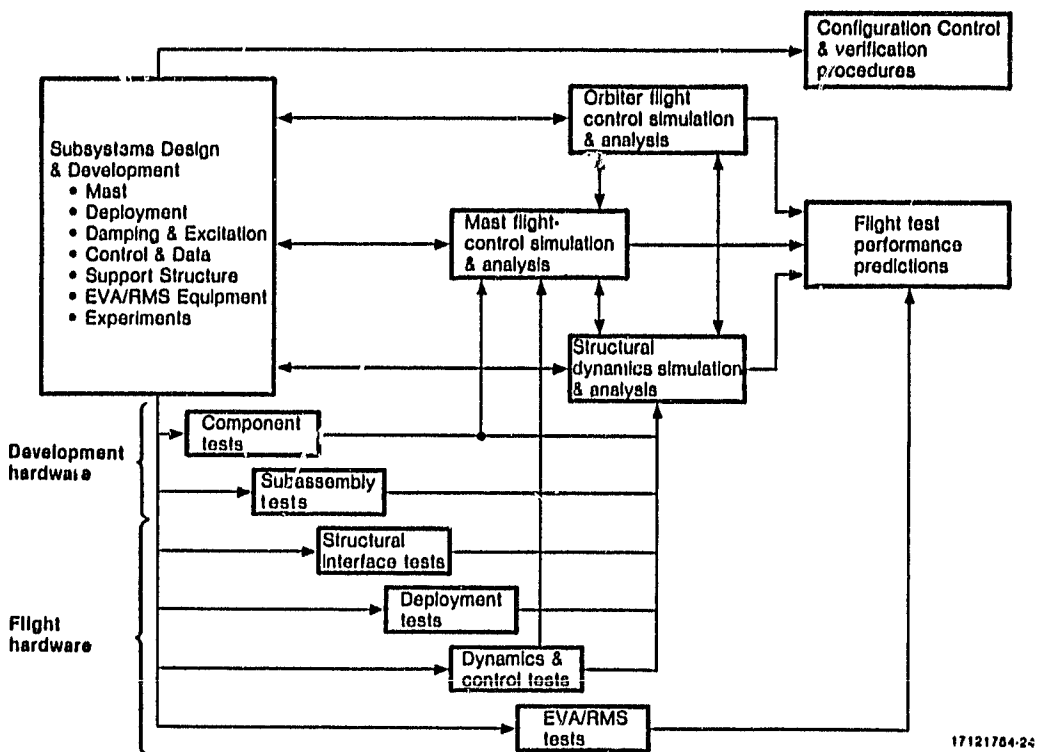
Figure 5-3. Ground Test and Simulation Approach
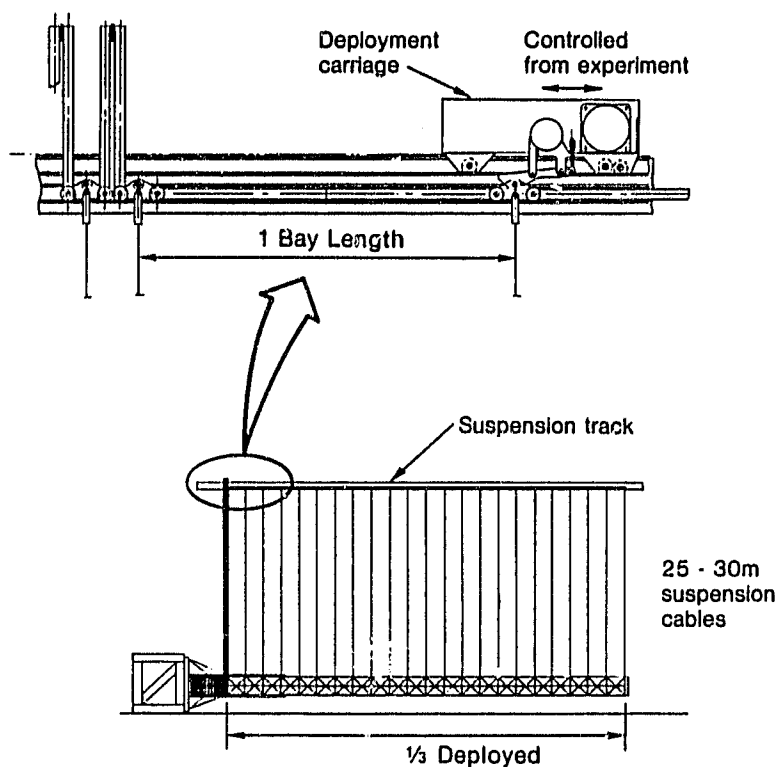
Figure 5-4. Ground Deployment Test Concept

The test will be performed on a 1/3 deployed truss, which is the initial deployed length that will be tested during flight test operations. A modal survey in the horizontal plane will be performed using the damper set torque wheels for excitation. Following excitation tests the damper sets will be activated and damping performance evaluated. The dynamics and controls tests will be performed in each of two planes by rotating the truss 90° about the longitudinal axis after the first test.

The dynamic model will include the suspension and gravity effects on the structure. Test results will be used to adjust the structural dynamic model as required to predict on-orbit dynamics.

5.7.3 STRUCTURAL INTERFACE TEST. The SCE support structure will be installed in a rigid test fixture with simulated Orbiter retention fittings to retain the structure at its five trunnion pins. A rigid load fixture will attach to the SCE support structure at all of the deployable truss attach points. The flexible base mount mechanism will be tested both in the locked out mode and the unlocked mode.

Force input and deflection will be measured at each of the truss attach points in real time along with the trunnion pin loads and motions while moments are applied to the load fixture about the pitch, yaw, and roll axis. The loads and deflections data will be used to generate a stiffness and/or flexibility matrix for the finite element simulation of the SCE.

5.7.4 EVA/RMS GROUND TESTS AND SIMULATIONS. The EVA and RMS ground tests and simulations will be conducted in two phases. One-g tests and simulations will be performed on the SCE installed in the JSC Manipulator Development Facility (MDF). Water bouyancy zero-g simulations and tests on a test segment of truss in the JSC Weightless Environment Training Facility (WETF).

5.7.4.1 MDF Testing. The objectives of the MDF one-g tests and simulations are as follows:

a. To verify the compatibility of the RMS/SCE deployment interfaces and actions.

b. To evaluate and refine EVA/RMS tasks and sequences in performing the flight test space construction experiments.

c. To evaluate special RMS tools and suitcase experiment hardware, human factors, and system compatibility.

d. To develop flight test procedures and initial timelines.

e. To train the SCE flight test personnel.

With the SCE installed in the MDF as shown in Figure 5-5, the flight test crew will perform the prescribed flight EVA/RMS test activities.
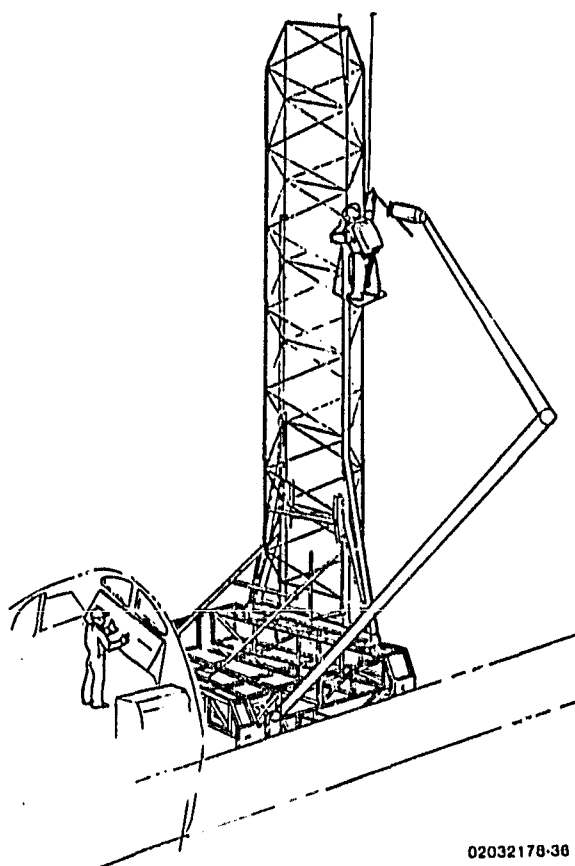
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Figure 5-5. MDF Tests and Simulations Concept

These include:

a. RMS aided deployment exercises

b. EVA test equipment unstowage and work station set-up

c. RMS-astronaut handoff and equipment transfer operations

d. Experiments installations

e. Experiments removal

f. Test equipment restowage

g. RMS/EVA aided truss restowage operations

h. Contingency emergency operations

The task sequences will be worked out and problems resolved through practice exercises on the partially deployed truss to determine the most effective and efficient procedures to follow.

5.7.4.2 <u>WETF Testing</u>. The objectives of WETF water bouyancy zero-g simulations and tests are as follows:

a. To evaluate the probable effects of zero-g on SCE flight test EVA personnel behavior, performance, and endurance for correlation with flight test results.

b. To evaluate potential space illumination and shadow effects on EVA personnel performance and establish illumination techniques and procedures for flight tests.

c. To determine water-space correlation factors and update flight test timelines.

The five-bay test truss segment used for the subassembly tests will be modified and installed in a support fixture to allow EVA work stations to be set up for the bouyancy tests as shown in Figure 5-6. The construction experiments will each be installed using a diver to simulate RMS handoffs to the crew member performing the tasks. The designated crew member and at least one back-up crew member will be evaluated. Alternative sequences and illumination techniques will be evaluated. Final procedures will be worked out and run through in sequence several times to acquire performance measures for correlation with flight tests. Performance evaluation will be based on crew member self-evaluation and observer-evaluations of recorded video tapes of test operations.
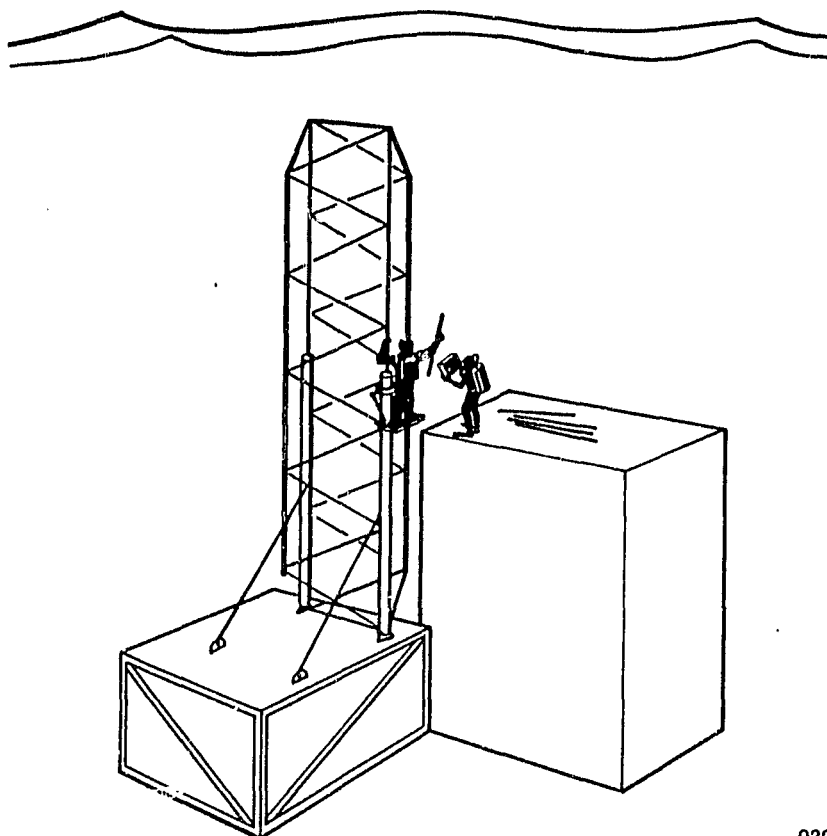
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Figure 5-6. WETF Simulations and Test Concept

5.8 FLIGHT CERTIFICATION TEST

The Flight Certification Test of the Space Construction Experiment will be performed at Johnson Space Center (JSC) in the test sequence as shown in Figure 5-7.

```
┌──────────────┐    ┌─────────┐    ┌──────────┐    ┌────────────┐    ┌────────────┐
│ EVA/RMS      │    │         │    │          │    │  RANDOM    │    │  THERMAL   │
│ &            │──▶ │  EMC    │──▶ │ ACOUSTIC │──▶ │ VIBRATION  │──▶ │  VACUUM    │
│ SIMULATION   │    │         │    │          │    │            │    │            │
└──────────────┘    └─────────┘    └──────────┘    └────────────┘    └────────────┘

JSC                                JSC              JSC                JSC
MANIPULATOR                        ACOUSTIC TEST    VIBRATION LAB      THERMAL
DEVELOPMENT                        FACILITY (GVL)   (GVL)              VACUUM LAB
FACILITY
(MDF)
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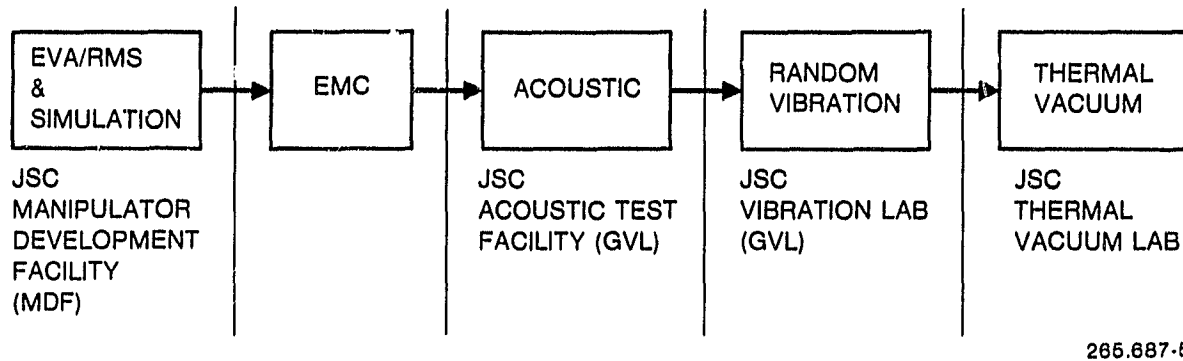
Figure 5-7. Flight Certification Test Sequence

The test objective is to demonstrate that the SCE including truss, cradle and stowed experiments will function satisfactorily after being subjected to Shuttle flight environments. The tests include the EMC test conducted at ambient conditions, followed by Acoustic test, Random Vibration test and Thermal Vacuum Test.

5.9 GROUND OPERATIONS PLAN

The general plan for SCE ground operations to be conducted at KSC during both preflight preparations for launch and subsequent postflight activities after landing is described in the following subsections.

5.9.1 PREFLIGHT GROUND OPERATIONS AT KSC. Initial preflight operations will be performed in a Payload Processing Facility (PPF) to be designated for SCE use. PPF tasks include receiving and inspection, refurbishment, preparation, and checkout operations as necessary to establish SCE system flight readiness.

The SCE will then be transferred to either a Vertical or Horizontal Processing Facility where it will be integrated with other assigned coflight manifested payloads (into a complete cargo assembly) and processed for launch using conventional Shuttle Orbiter preflight procedures. Either the vertical or the horizontal processing mode may be used for the SCE, permitting flexibility in its selection for compatibility with other payloads. Although basically the same operations are performed in either mode, each is discussed separately because different facilities/procedures are used in each.

5.9.1.1 Payload Processing Facility (PPF) Operations. A sequence diagram of the operations to be performed in the PPF are presented in Figure 5-8. These activities, encompassing approximately six weeks of SCE preparation and check-out tasks, are described below.

a. Following flight certification testing at JSC, the SCE will be shipped to KSC.

b. Upon arrival at the designated KSC PPF, the SCE equipment will be unpackaged. An initial inspection will then be performed.

c. Other items to be received and inspected in the PPF will include the flight instrumentation components (strain gages, thermocouples, accelerometers, and load cells) and associated cabling, and a simple ground test switch panel.

d. The truss assembly will be deployed horizontally while installed in its handling and transportation dolly. The ground test switch panel and a power supply (simulating the Orbiter 28 vdc power) will be connected to the SCE and a preliminary electrical check performed. In preparation for truss extension, the truss sidemembers will be manually unlatched and positioned.
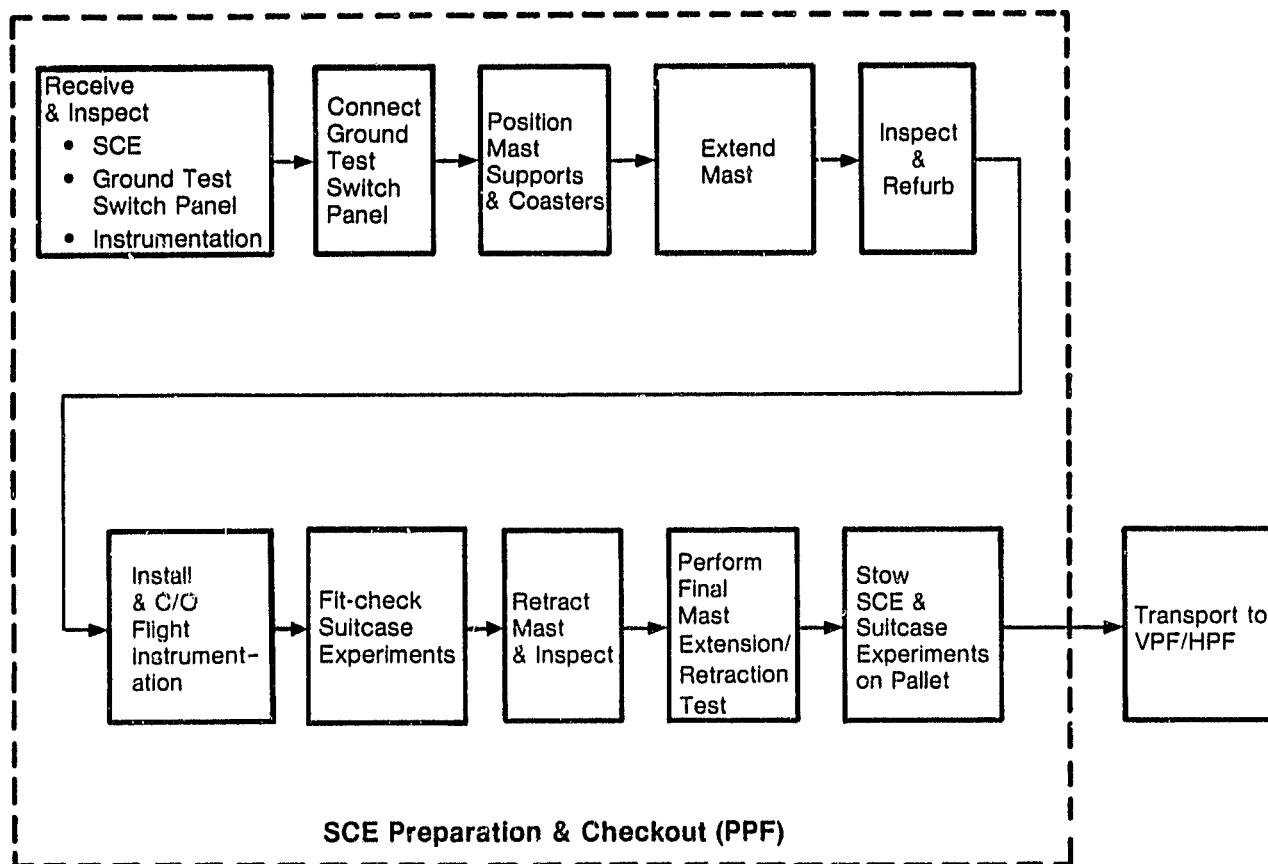
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Figure 5-8. SCE Payload Processing Facility Operations

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e. The truss will then be fully extended (in increments of several bay groups at a time) as commanded from the switch panel operating the SCE control unit (CU). As the truss is extended, GSE support dollies will be manually positioned under the structure to provide physical support in the extended configuration and to allow the necessary movement of the truss across the floor.

f. In the fully extended position, a complete inspection of the mast structure will be accomplished and any discrepant areas refurbished.

g. Flight instrumentation, electrical equipment, and harnessing will then be installed on the SCE and applicable functional checks and calibrations performed. The suitcase experiment(s) will also be checked out and installation capabilities verified. Other checks will include an end-to-end test of the tip mass jettisoning system.

h. The mast will then be retracted (using the ground test switch panel and SCE CU for control), and an inspection performed in the retracted configuration. This will be followed by a final extend/retract cycle to verify that the added instrumentation components and harnessing do not adversely do not adversely affect the deployment and retraction processes. During this final cycle, prior to retract, a complete cleaning of the mast structure will be performed.

i. The truss will be fully retracted and folded to its stowed configuration. The truss will be electrically disconnected and lifted by handling sling from its dolly. The truss will be installed vertically on the FSE support structure which will be mounted on its handling and transportation trailer.

j. After installation of the pitch braces, all power, data and control harness connectors will be mated and all circuits functionally checked. The truss will be rotated from the vertical to horizontal position to vertical position several times to verify no interference with harnesses exists and to test the pitch strut and holddown latches. The spring mount latches will be cycled to verify their function.

k. All suitcase experiment hardware will be secured in the stowed positions on the support structure and the truss rotated to the stowed position and latched. The SCE will then be prepared for transportation to either the Vertical Processing Facility (VPF) or the Operations and Checkout Facility (O&C) which would be the Horizontal Processing Facility (HPF). The subsequent preflight operations are summarized in Figure 5-9 and described in the following subsections.

5.9.1.2 <u>SCE Vertical Processing Operations</u>. In the vertical processing mode, preflight operations will be performed at three separate facilities: the PPF (previously discussed above), the VPF, and the launch pad. The general flow sequence of operations to be performed in each of these facilities for vertical
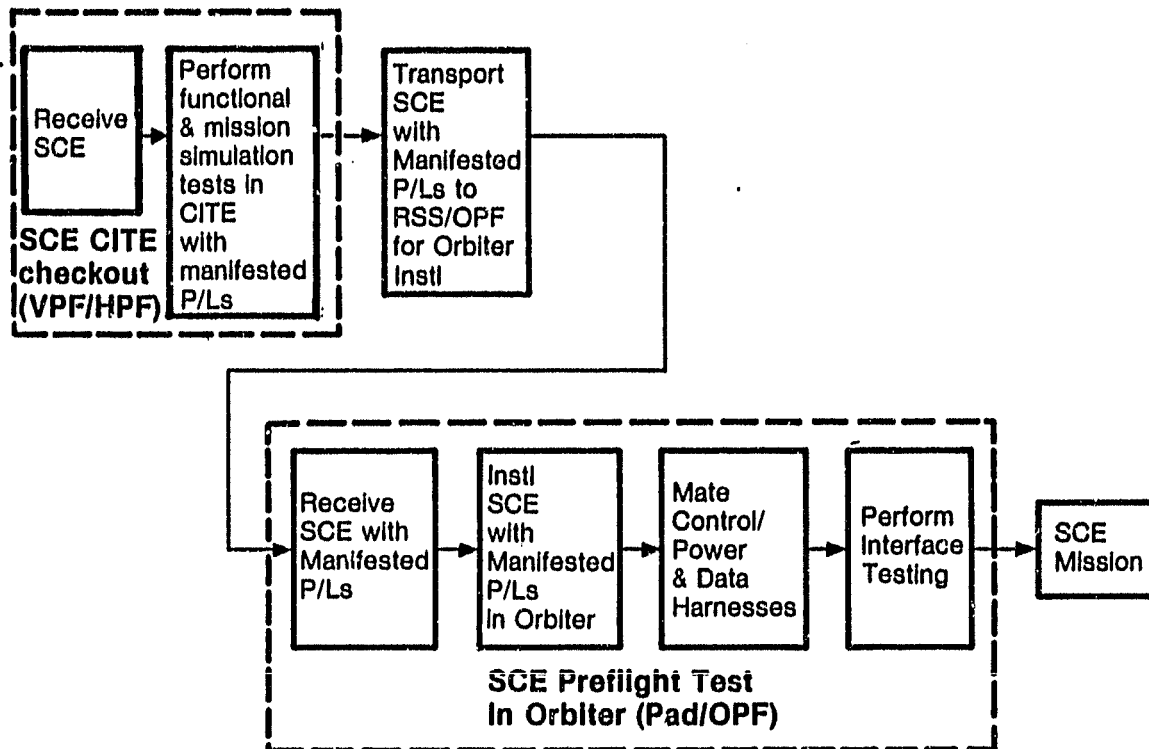
Figure 5-9. SCE Preflight Ground Operations Sequence Summary   17121764-31

processing of the SCE is depicted in Figure 5-10. Timespan requirements for the major activities involved are shown in Figure 5-11. Further description of the VPF and launch pad operations is provided below.

Upon arrival at the VPF, the SCE will be removed from its handling pallet and placed in the Vertical Payload Handling Device (VPHD) where it will be physically integrated with its other coflight manifested payloads. The SCE (and the coflight payloads) will then be connected to the Cargo Integration Test Equipment (CITE) which electrically simulates the flight Orbiter. The Orbiter standard switch panel (or its equivalent) to be used for SCE control is provided in the simulated Aft Flight Deck, and all interface cabling will be installed within the test stand as appropriate.

Following preliminary interface tests, approximately three and one-half weeks of integrated CITE testing with the manifested payloads will be performed. The SCE portion of these CITE tests will consist primarily of functional and mission simulation tests.

After completion of CITE testing, the SCE and manifested payloads will be placed into the Multiuse Mission Support Equipment (MMSE) canister and transferred to the launch pad aboard the MMSE transporter.

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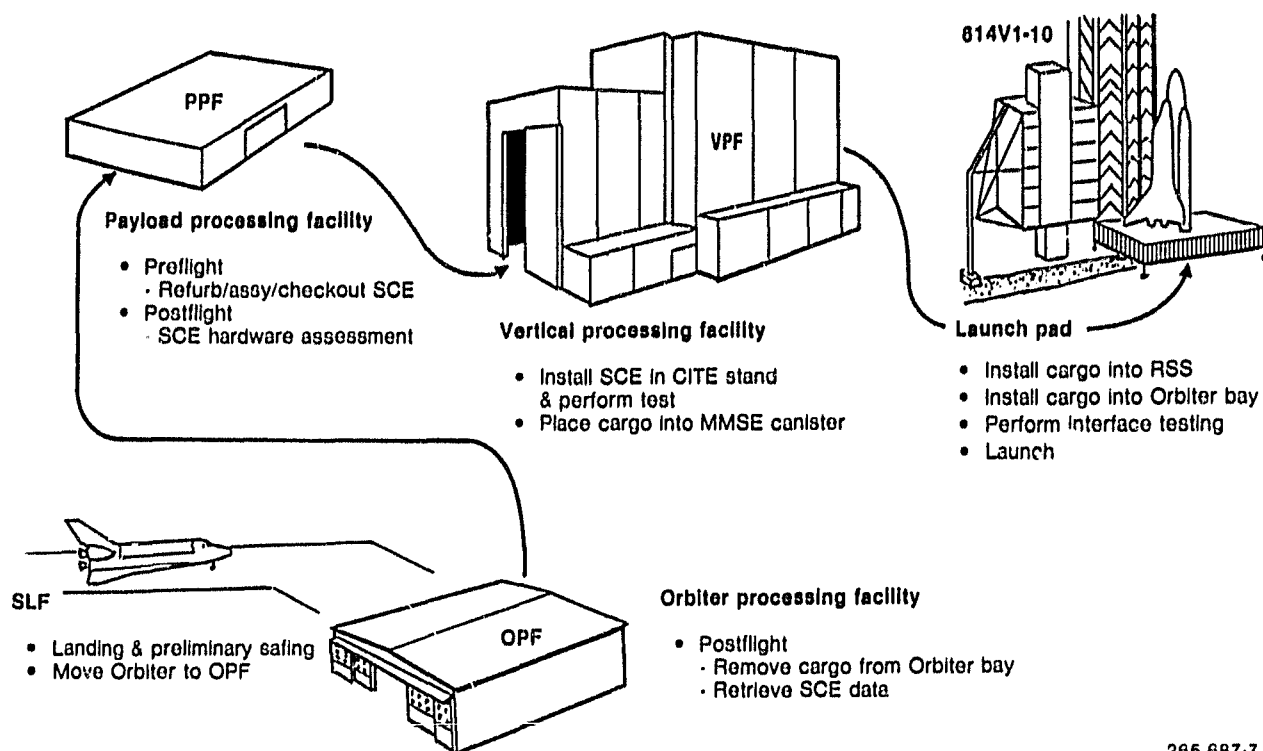
Figure 5-10. SCE Vertical Processing Operations

*Based on STS-5 timelines dated Oct 81
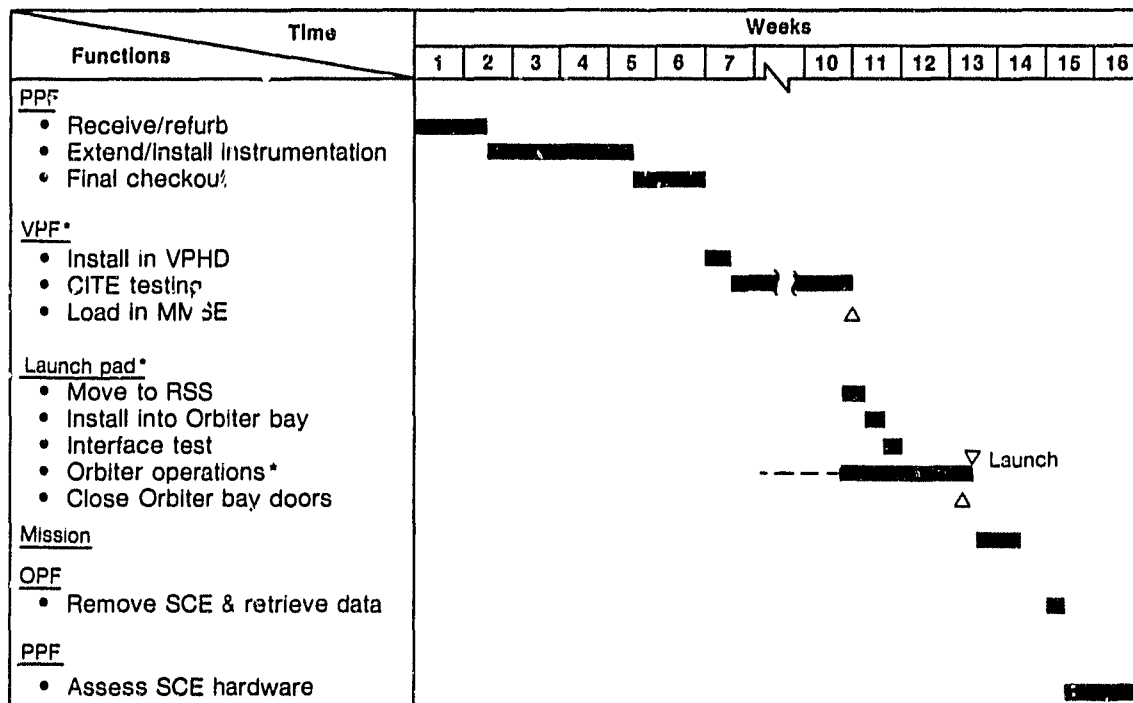
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Figure 5-11. SCE Vertical Processing Timeline

At the launch pad, the payloads will first be placed in the Rotating Service Structure (RSS) which in turn will be used to install the payload into the Orbiter bay. After physical installation is complete, all SCE/Orbiter interface harnesses will be connected.

A series of brief interface checks will then be performed to verify all SCE power, control, and data circuits. From this point on through launch and up until SCE mission deployment, the SCE is essentially dormant except for final pyrotechnic bolt installation and connections.

After completion of approximately one additional week of Orbiter checkout operations, the Orbiter and its payload are ready for launch.

5.9.1.3 SCE Horizongal Processing Operations. In the horizontal processing mode, the SCE will be cycled through five separate facilities during preflight operations: the PPF (discussed previously), the O&C (which acts as the horizontal processing facility), the OPF, the VAB, and the launch pad. The general flow sequence of operations through these five facilities is illustrated in Figure 5-12. Timespan requirements for the major activities involved are shown in Figure 5-13. Description of the O&C, OPF, VAB and launch pad operations are provided below.

Following checkout in the PPF, the SCE will be transferred to the O&C facility for horizontal processing. The operations to be performed in the O&C are virtually the same as those performed in the VPF except they are conducted with the SCE (and other coflight payloads) oriented in a horizontal rather than vertical attitude.

Upon arrival in the O&C, the SCE will be placed in a horizontal test stand and integrated with its other coflight payloads. The Cargo Integration Test Equipment (CITE) will then be connected to the SCE, followed by integrated CITE testing with the other manifested payloads. The SCE portion of these CITE tests will consist of functional and mission simulation tests.

After completion of CITE testing, the SCE and manifested payloads will be placed into the MMSE canister and transferred to the OPF.

At the OPF, the SCE and coflight payloads will be installed in the Orbiter cargo bay. After physical installation is complete, all SCE/Orbiter interface harnesses will be connected.

A series of brief interface checks will then performed to verify all SCE power, control, and data circuits. From this point on through launch and up until SCE mission deployment, the SCE is essentially dormant. No further access to the SCE is required.

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Payload processing facility

- Preflight
  - Refurb/assy/checkout SCE
- Postflight
  - SCE hardware assessment

**Operations & checkout
(Horizontal processing facility)**

- Install SCE in CITE stand
  & perform test
- Place cargo into MMSE canister

Orbiter processing facility

- Preflight
  - Install cargo into Orbiter bay
  - Perform interface testing
- Postflight
  - Remove cargo from Orbiter bay
  - Retrieve SCE data

Vertical assembly building
- No SCE tasks

SLF
- Landing & preliminary safing
- Move Orbiter to OPF

Launch pad
- Launch

Figure 5-12.   SCE Horizontal Processing Operations   265 687-9

Functions (Time → Weeks)	1	2	3	4	5	6	7		10	11	12	13	14	15	16	17
PPF																
• Receive/refurb	▬															
• Extend/install instrumentation		▬▬														
• Final checkout			▬													
O&C*																
• Transfer to O&C						▪										
• CITE testing							▬									
• Load in MMSE							▪									
OPF*																
• Install in Orbiter bay							▪									
• Interface testing							▪									
• Close Orbiter bay doors							△									
VAB*								▬								
PAD*																
• Orbiter operations thru launch*								▬	▽							
MISSION										▬						
OPF																
• Remove SCE & retrieve data															▪	
PPF																
• Assess SCE hardware																▬
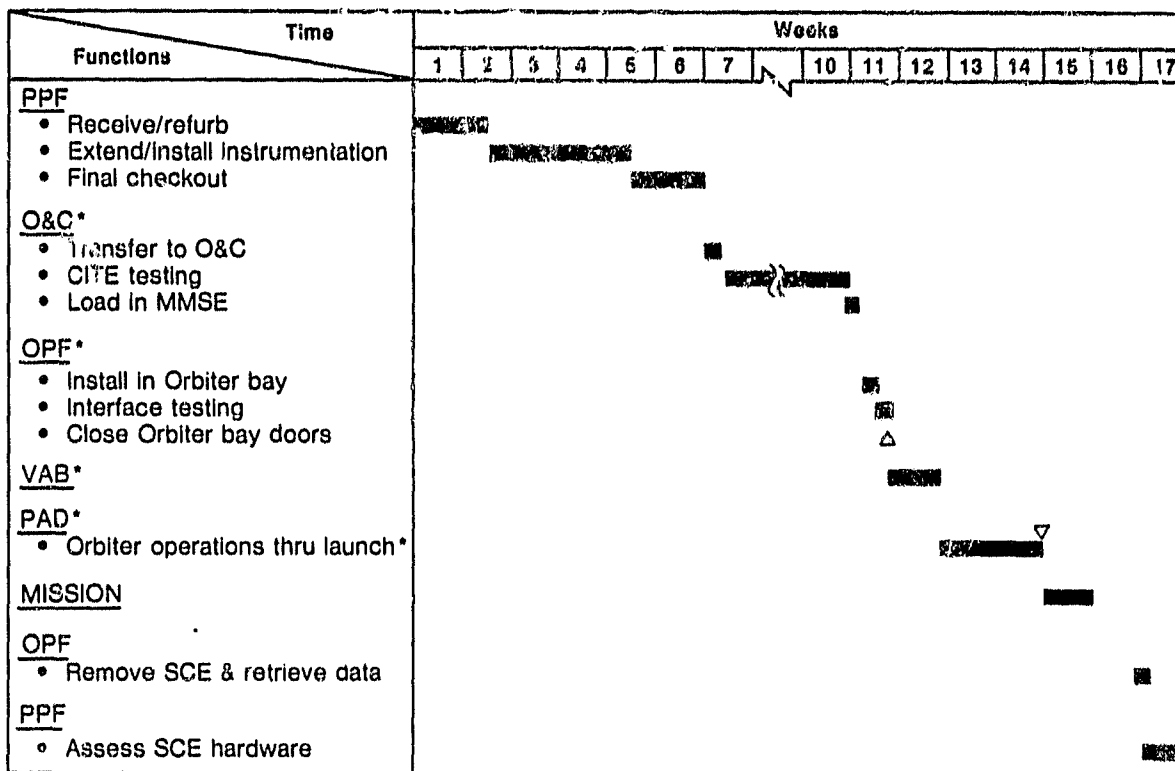
*Based on STS-5 timelines dated Oct 81   265 687-10

Figure 5-13.   SCE Horizontal Processing Timeline

Following these interface checks, the Orbiter cargo bay doors are closed and the Orbiter will be towed to the VAB.

In the VAB, the Orbiter will be erected to a vertical attitude and mated to the external tank and solid rocket boosters (SRB's) on the Mobile Launch Platform (MLP). These operations involve approximately one week of space shuttle activities only; no SCE operations are required.

After completion of the VAB operations, the entire vehicle assembly (with the SCE installed in the Orbiter cargo bay) will be transported to the launch pad and prepared for launch. These operations require approximately three weeks of space shuttle activities. Final SCE operations require installation of pyrotechnic bolts in the tip mass ejection mechanism.

5.9.2 <u>POSTFLIGHT GROUND OPERATIONS AT KSC</u>. Following completion of the SCE flight mission, the SCE will be returned to KSC by the Orbiter. The postflight operations required by the SCE at KSC are described below. A block diagram of these operations is shown in Figure 5-14.

After the mission is completed and the Orbiter has landed, it will enter the OPF. The flight data recorder tapes will be removed from the Orbiter. The SCE will be lifted out of the Orbiter bay using the MMSE strongback and placed on the shipping/handling trailer. The SCE will be transported to the PPF.
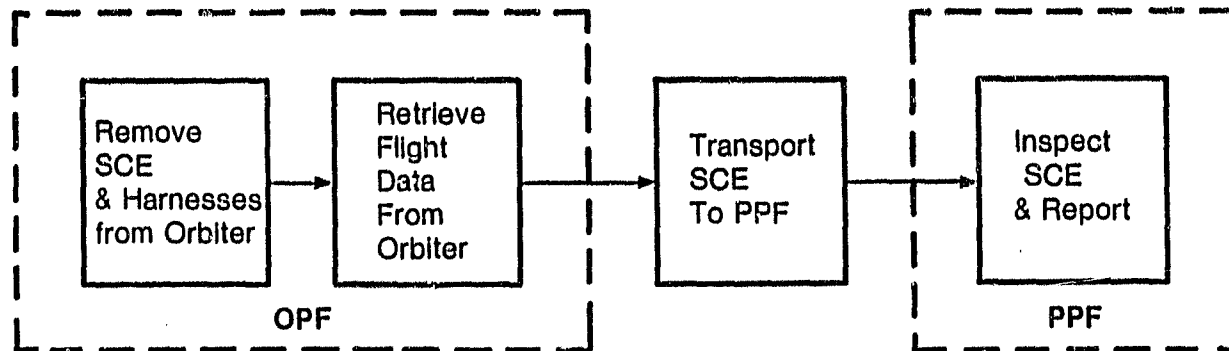
Figure 5-14. SCE Postflight Ground Operations Sequence

The truss assembly will be removed from the ASE support structure and installed on its handling and transportation dolly. The truss will be electrically connected to the support structure subsystems. The ground test switch panel and power supply will be connected to the CU and the truss will be fully deployed on its support dollies.

The structures and components will be inspected for evidence of damage and degradation. All discrepancies will be documented.

Following the inspections the truss will be repackaged and prepared for final disposition.

5.9.3 **GROUND SUPPORT EQUIPMENT (GSE) REQUIREMENTS.** GSE items required to support SCE preflight and postflight ground operations are listed in Table 5-3.

Table 5-3. GSE Items for the SCE

Item	Quantity	Purpose
Ground Test Switch Panel	1	Checkout, deployment & retraction control
Truss Handling & Transportation Dolly	1	Ground handling and transport of truss assembly
Payload Handling & Transportation Trailer	1	Ground handling & transport of ASE
Truss Support Dollies	20	Support truss during ground deployment
Payload Handling Sling	1	Pick-up ASE support structure or fully assembled payload
Truss Handling Sling	1	Pick-up truss assembly
Cable Kit	1	Interconnect power, data and control functions for ground test and checkout.

5.10 FLIGHT OPERATIONS PLAN

The flight test sequence will require two days of the total mission. The first few days in orbit will be used to deploy the satellite payloads. Following these operations the SCE activities will be initiated.

The flight test operations sequence and timelines for the first day of the experiment are shown in Figure 5-15. The first day's activities include the series of controls and dynamics tests described in Figure 5-16.
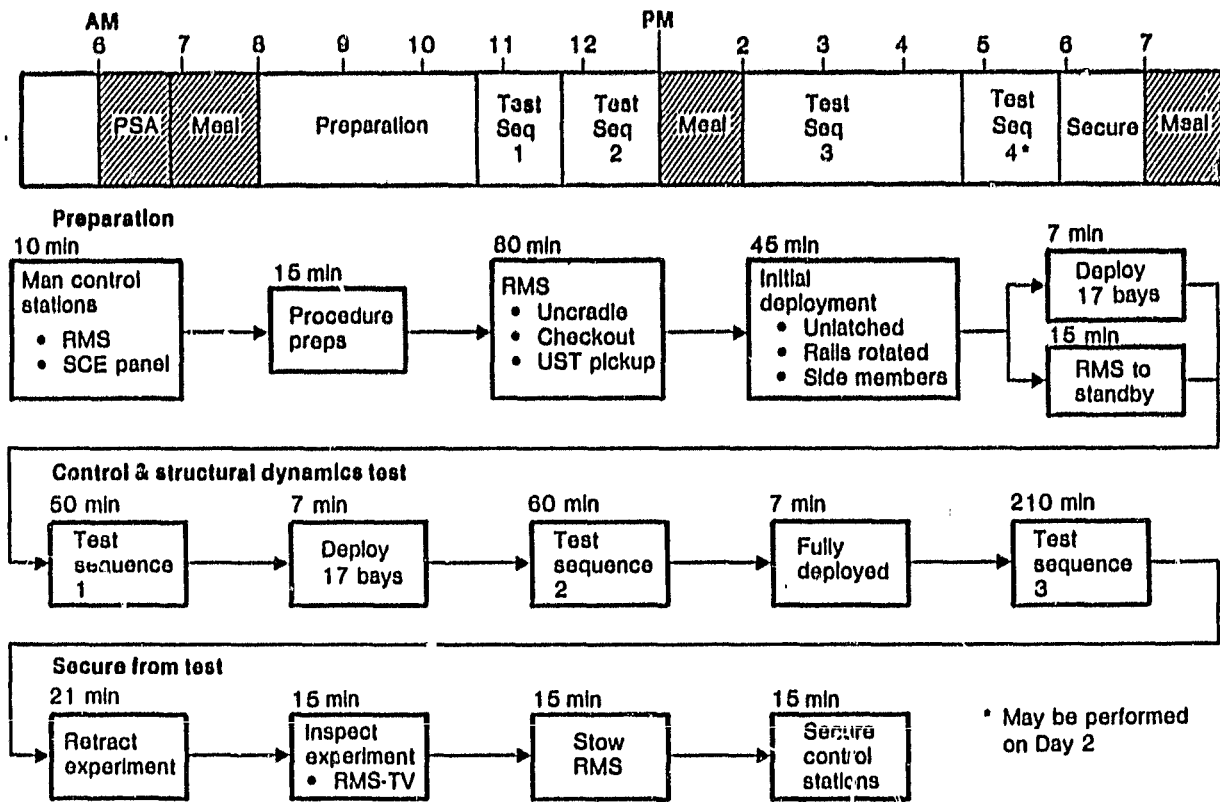
Figure 5-15. Flight Test Operations Sequence and Timelines for Day 1
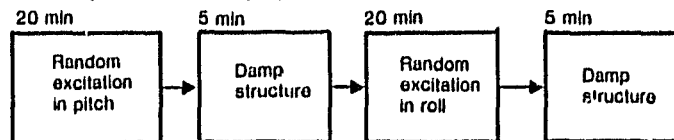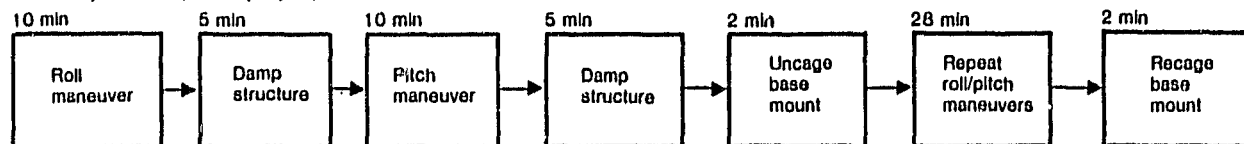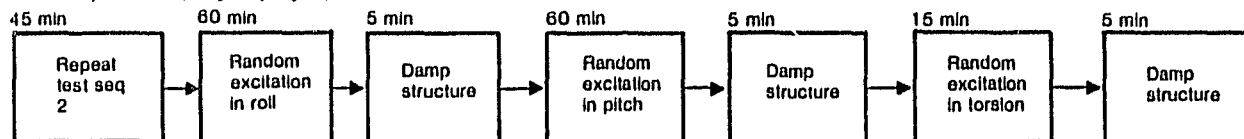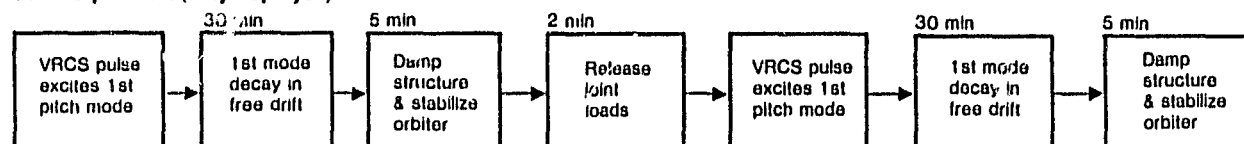
Figure 5-16. Structural Dynamics and Controls Flight Test Sequence

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The day one experiments will be conducted by the payload specialist and the pilot and/or mission specialist working at the aft flight deck control and display panels. The payload specialist will control and monitor the experiment while the pilot performs RMS operations and controls the roll and pitch maneuvers.

The dynamics and controls flight test sequences are outlined in Figure 5-16. Test sequence 1 of the 1/3 deployed structure performs modal surveys in pitch and roll with intermediate and final damping operations to stabilize the structure prior to the start of each test. The 1/3 deployed configuration corresponds to the configuration tested as part of the ground test program so that comparative data are obtained.

Test sequence 2 is the first test of the effects of the structural inertia and low frequency on the Orbiter and DAP. This series of tests would be performed using appropriate combinations of rate limit and pointing limit. This sequence and the first part of test sequence 2 will allow the limits of DAP control to be approached using decreasing steps of low frequency with increased steps in moments of inertia.

Test sequence 3 includes extended random excitation in order to provide data on a wide range of modes. Excitation and measurement of torsional modes are also included.

Test sequence 4 uses the RCS to excite the first mode in pitch with subsequent measurement of first mode decay in free drift. This test is run with and without joints unloaded to establish damping characteristics and the effects of joint clearance on damping. Test sequence number 4 could be performed early on the second day of the experiment prior to the EVA egress. This would provide more contingency time for the day 1 activities.

The construction operations test sequence will be conducted on the second day of the experiment. This test sequence, illustrated in Figure 5-17, includes several assembly and installation tasks that require manual and EVA-assisted operations. The EVA tasks will be performed by the mission specialist and the commander. The payload specialist will continue to control and monitor the SCE from the aft flight deck control and display panel, while the pilot performs the RMS operations. The EVA will remain in effect until all equipment is fully stowed for reentry and landing.

5.11 INSTRUMENTATION PLAN

Instrumentation requirements for the SCE are summarized in Table 5-4.
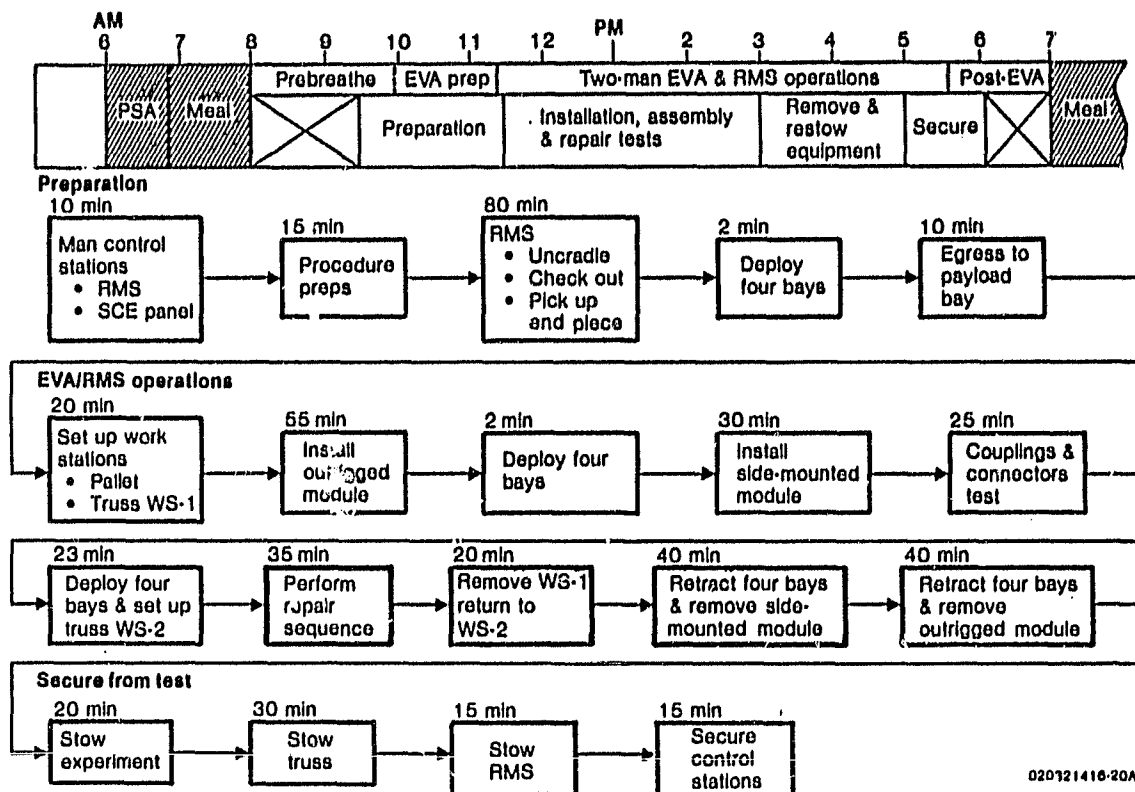
Figure 5-17. Construction Operations Test Sequence and Timelines for Day 2

Table 5-4. SCE Instrumentation Requirements

No.	Measurement	Type Sensor	Qty	Location
1	Tip motion rate	Rate gyro	6	1 each damper set
2	Mode shape & frequency	Servo-accelerometer	18	3 each at 6 truss stations
		Rate gyro	3	1 each at 3 truss stations
		P/E accelerometer	12	3 each at 4 truss stations
3	Z-axis acceleration	P/E accelerometer	1	Tip of truss
4	Tip deflection	Laser & detector array	1	Tip & base of truss
5	Carriage position	Rotary encoder	2	1 each deploy carriage
6	Motor temperatures	Thermocouple	10	2 each carriage 1 each damper set
7	Truss member load	Strain gauge	48	2 each longitudinal & diagonal, truss bay 33 & 50
8	Roll support loads	Strain gauge	4	1 each deployment rail Roll support lug
9	Pitch support loads	Load cell	2	1 each pitch brace
10	Trunion pin loads	Strain gauge	10	2 each pin
11	Trunion pin motions	Potentiometer	5	1 each pin
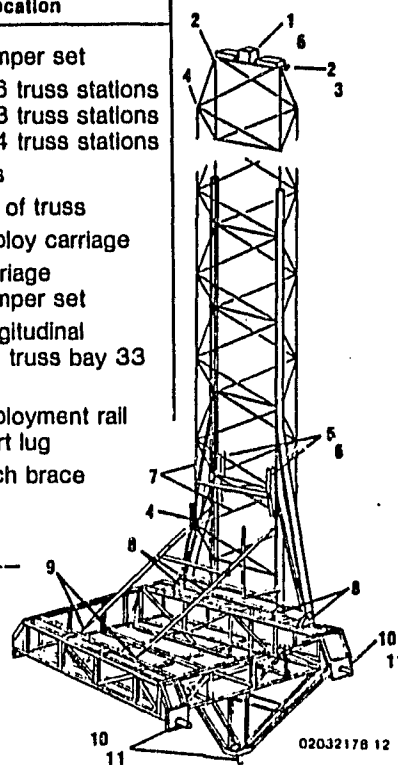
5.12 OPTIONAL FLIGHT EXPERIMENT

5.12.1 RCS PLUME EFFECTS EXPERIMENT. It was suggested by the NASA that the deployable truss could possibly be used to support sensors to measure RCS plume effects at several distances from the Orbiter. The concept shown in Figure 5-18 would place sensors at the points where the plume core from fore and aft thrusters intersect the truss. A third sensor at 90 meters from the Orbiter would see either or both plumes.

The plume characteristics of interest for these types of experiments are the plume pressure, condensible volatiles, particulates, and high velocity ice crystals. The concers are the potential damage and contamination of highly sensitive spacecraft instruments and arrays by the Orbiter RCS plumes.

The candidate sensors to consider for performing plume effects measurements are described in Table 5-5.

Sensor locations
- A — Forward plume core intersects at $\simeq 20$ meters
- B — Aft plume core intersects at $\simeq 38$ meters
- C — $1/R^2$ both plumes confirmed at >90 meters

RCS plume characteristics
- 38 RCSs   875 lb
- 2,140 gms/sec/per RCS
- 17½ deg ½ angle core
- Constituents

H_2O	32%
Lightweights — H, N, CCT	$>60\%$
Nitrates	$\simeq 2$-3%

- Velocity $\approx 3,000$ m/sec
- Ice crystal size ≈ 400-1,000 microns
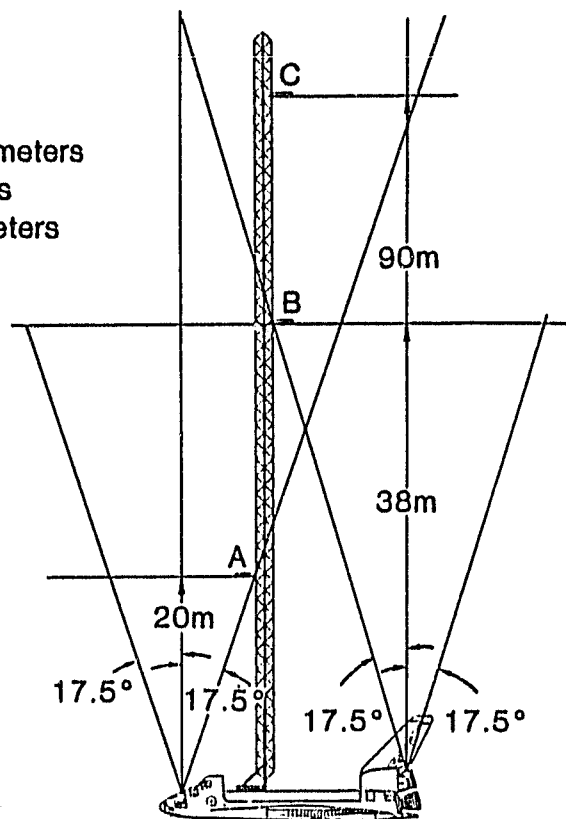
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Figure 5-18. RCS Plume Effects Experiment Concept

Table 5-5. Plume Effects Sensors to Consider

Baratron
- Measures plume pressure
- Now a part of IECM
- Thin film could be damaged by liftoff or particulate

TQCM
- Thermoelectric quartz crystal microbalance
- Measure condensable volatiles
- Now a part of IECM
- Minimum temp of $-60°C$ will not condense H_2O
- Relatively high power
- Away from RCS nitrates are probably atomized & collectible
- Could be used for non H_2 collection (i.e., the bad stuff)

Mass spectrometer
- Measures & classifies the complete plume
- University of Michigan version on IECM
- Expensive
- Could be used for $1/R^2$ validation

Heat flux calorimeter
- Measures degradation of α/ϵ due to contamination
- Solar flux heats black surface & surfaces of materials of interest
- Relative surface temperature measurement
- JPL design used on NOAA & NOVA

TEOM
- Tapered element oscillating microbalance
- Particulate trapper

Impingement sensor
- Acoustical sensor
- Like micrometeorite detector
- Measure accelerated ice particles
- Classify by momentum

Surface damage
- Passive array — "Measure Impact of Impact"
- Solar cell cover glass
- Silver coated teflon thermal surfaces
- Thin foils
- Bring back for damage study
- Various distances in plume core to determine
  minimum operating range of RCSs in docking maneuver

Examples of the available options that might be performed in conjunction with the SCE are given below with their approximate costs. A more detailed study is required to establish installation techniques for these sensors and to determine the length of time and RCS activity required to conduct such an experiment.

a. Option 1

For < $200K + integration cost

- Passive surface damage

- Heat flux calorimeter

- TQCM

b. Option 2

For approximately $1M + integration cost

- Impact sensor + passive array surface damage

- Combined heat flux calorimeter and TQCM

- Plume pressure experiment

SECTION 6

PRELIMINARY PROGRAM PLAN

6.1 COST ANALYSIS

Preliminary ROM cost estimates have been prepared for the candidate Space
Construction Experiment (SCE) concept described in this report. Two annual
funding requirements have also been developed in accordance with the nominal
and alternate program schedule options discussed in Section 6.2 .

6.1.1 <u>METHODOLOGY</u>. The parametric cost model used for this analysis is
an adaptation of our Space System Life Cycle Cost (SSLCC) model tailored
specifically for the SCE. The SSLCC model was developed in-house over the
last several years and used extensively for the SCAFEDS, Geostationary Plat-
form Study, OTV study, and other studies of similar flight vehicles.

Initially a cost-related work breakdown structure (WBS) was developed that
included all elements incurred by the SCE project for each program phase:
development, production, and operations. Operations costs are not addressed
in this study. This cost WBS then sets the format for the estimating model,
the individual cost estimating relationships (CERs), cost factors, or specific
point estimate requirements, and the cost estimate output. Estimates are
then made for each cost element either at the breakdown level shown or, in
certain cases, one level lower. These estimates are then accumulated to pro-
vide the cost for each program phase.

The estimating methodology varies with the cost element and with the availability
of historical data or supplier estimates. Where sufficient detailed definition
of the hardware and tasks are available, detailed estimates of labor and material
may be developed. This procedure was used to develop the cost of the deploy-
able truss beam. Drawings, parts lists, and fabrication description were used
to generate material procurement requirements and labor hours for design and
analysis, tooling design and fabrication, test article manufacturing, development
test, GSE design and fabrication, sustaining engineering and tooling, acceptance
test, and quality assurance. These labor and material requirements are then
translated into dollar projections.

For other new hardware, parametric CERs are used. These CERs have been
derived for various families of hardware and many subcategories, representing
differing levels of complexity. They are derived from available historical cost
data or detailed estimating information and relate cost to a specific driving param-
eter such as weight, area, power output, etc. For example, the various experi-
ment structural items (other than the truss beam) were estimated using CERs.

Engineering point estimates were used for specific pieces of known equipment where the definition data were sufficiently detailed or the hardware item was existing equipment and cost data were available; for example, ROM estimates for some of the dynamic test equipment (gyros, etc.).

The remaining wraparound cost elements, such as system engineering and integration, program management, etc., are estimated using cost factors consisting of appropriate percentages of the applicable related program effort.

The nonrecurring or development phase includes all one-time tasks and hardware required to design and test the equipment. It includes the design and analysis of all ground and flight hardware including structural analysis, stress, dynamics, thermal, mass properties, etc. The nonrecurring category also includes all component development and test through component qualification as well as all component development test hardware. In addition, this phase includes: software development; system engineering and integration; system level test hardware and the engineering test prototype and qualification article; and system test. Since the prototype approach will be used for this experiment, a single flight article will be manufactured and all system level testing will be accomplished using the flight vehicle, which will then be refurbished and updated to flight configuration. Also included in this phase are GSE design, development, test, and manufacture; facilities; and overall program management and administration.

The production phase (unit cost estimate) includes all tasks and hardware necessary to fabricate one complete set of flight hardware equipment. It includes all material and component procurement, parts fabrication, subassembly, and final assembly. In addition, this category includes the required quality control/inspection task, an acceptance test procedure for sell-off to the customer, and program management and administration activities accomplished during the manufacturing phase.

Operating costs, NASA ground testing, and Shuttle-user charges were not included in the cost analysis at this time.

6.1.2 GROUND RULES AND ASSUMPTIONS. The general ground rules and assumptions governing the subsequent cost estimates are:

a. Costs are estimated in constant 1982 dollars.

b. Prime contractor fee is not included.

c. Costs are for the design, development, and fabrication of a single, flyable experiment.

d. All system testing required is accomplished using the flight article hardware which is then refurbished for flight.

e. System testing conducted by NASA is excluded.

f. No mission operations or Shuttle-user charges are presently included.

g. The cost estimates presented are rough-order-of-magnitude costs, for planning purposes only.

6.1.3 WORK BREAKDOWN STRUCTURE (WBS). The WBS is a comprehensive breakdown of all program life cycle elements, categorized or sorted into several levels of hardware and task or function-oriented end items, and serves to identify the cost elements to be included in the cost analysis task. This WBS contains all hardware and tasks associated with Phase C/D development and test, fabrication of flight hardware, and the activities incurred during the test flight. It serves as the basic format for cost reporting and programmatic data, and to organize, plan, and manage the subsequent program. The WBS developed for the SCE is shown graphically in Figure 6-1, and each element is briefly defined below.
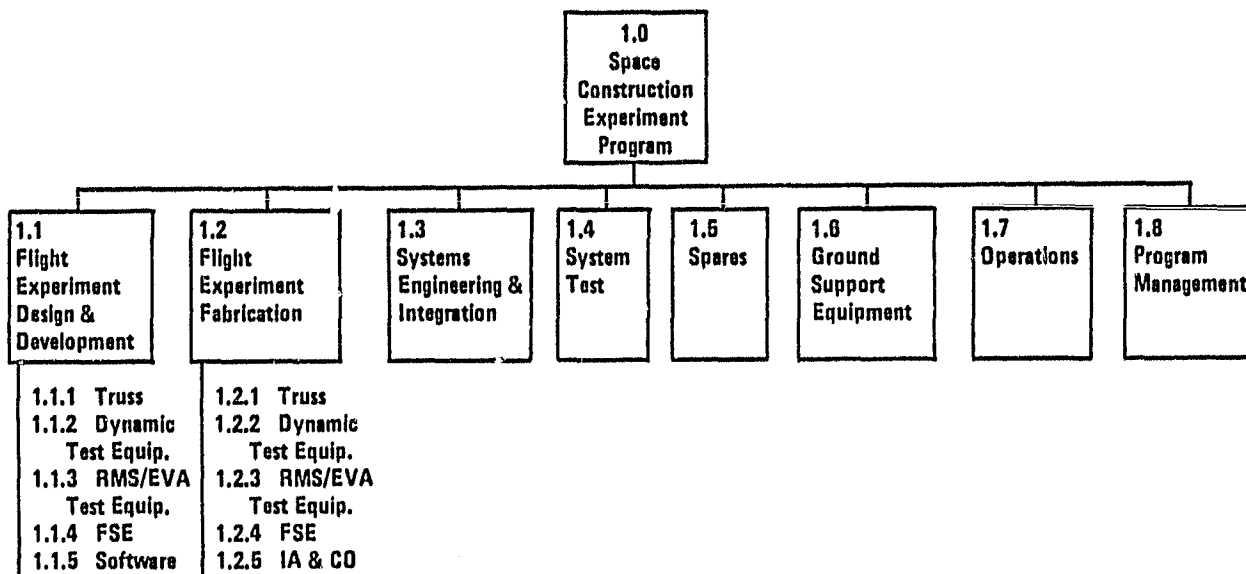
Figure 6-1. Space Construction Experiment WBS

a. WBS 1.0 - Space Construction Experiment Program. This WBS element summarizes all effort and material required for the design, development, fabrication, assembly, test and checkout, and operation of the SCE.

b. WBS 1.1 - Flight Experiment Design and Development. The design and development activities include all tasks and hardware for design and development and testing of the SCE. It includes the required design and analysis for all ground and flight hardware, including structural analysis, stress, dynamics, thermal, mass properties, etc. This nonrecurring category includes tooling, component development, and test through component qualification, as well as all component development test hardware. This element also includes software development.

c. WBS 1.1.1 - Truss. The deployable truss is the primary structural element being tested. It has a diamond cross section and 50 bays and is constructed of composite materials. Also included are the deployment mechanism, experiment support elements, and the tip mass.

d. WBS 1.1.2 - Dynamic Test Equipment. The equipment includes torque wheels and torque motor controllers, gyros, accelerometers, loads, displacement and temperature instrumentation, and their wiring harness. This equipment excites and measures vibrational modes and system parameters and provides active damping augmentation.

e. WBS 1.1.3 - RMS/EVA Test Equipment. The RMS/EVA test equipment includes dummy "black boxes" and attachment fittings, a dummy cabling harness and attach fittings, a portable EVA workstand, and special RMS end pieces.

f. WBS 1.1.4 - Flight Support Equipment (FSE). The FSE consists of the equipment supporting structure, a data acquisition system and a control unit, and their wiring harnesses.

g. WBS 1.1.5 - Software. This WBS element consists of all labor, material, and computer resources necessary to provide validated SCE flight software. It includes the design, programming, validation, and verification.

h. WBS 1.2 - Flight Experiment Fabrication. The flight experiment fabrication cost element includes all tasks and hardware necessary to provide one complete set of flight hardware equipment. It includes all material and component procurement, parts fabrication, subassembly, and final assembly. In addition, this category includes the required quality control/inspection task, an acceptance test procedure for sell-off to the customer, and program management and administration activities accomplished during the manufacturing phase.

i. WBS 1.2.1 thru WBS 1.2.4 - Subsystems. See above.

j. WBS 1.2.5 - Integration Assembly and Checkout. This WBS element consists of all effort and materials required to accomplish subsystem installation, final assembly, checkout, and acceptance testing of all mission payloads carried on the platform. These are all ground activities and culminate in sell-off to the NASA (DD Form 250).

k. WBS 1.3 - Systems Engineering and Integration. This WBS element summarizes all system level studies, analyses, and tradeoffs to support the development of requirements, specification, and interfaces necessary to direct and control the design of the overall system. It also includes all mission studies and analyses to establish requirements and planning for all phases of the mission and logistics activities. It also includes all product assurance activities consisting of safety, reliability, maintainability quality assurance, and parts, material, processes (PMP) control.

1. WBS 1.4 - System Test. This WBS element summarizes all effort and hardware required to conduct and support all major system level testing conducted by the contractor necessary to refine and validate the design and verify the accomplishment of the development requirements. They may include but not be limited to full-scale structural tests, integrated avionics tests, all-up functional tests, and payload functional and integration testing. This element includes test article refurbishment and reconfiguration; test planning, test analysis, preparation, and test operations; as well as test software and test support activities performed prior to delivery to the NASA.

m. WBS 1.5 - Spares. Ths WBS element includes the procurement and/or fabrication of all spare and repair parts necessary for the develoment and operational period.

n. WBS 1.6 - Ground Support Equipment (GSE). This WBS element summarizes all effort and material required to define, design, develop, test and qualify, procure, fabricate, assemble, and checkout all GSE required to support the SCE during the development, manufacturing, and operations phase. It includes all necessary handling and transportation equipment, and functional checkout equipment.

o. WBS 1.7 - Operations. This WBS element summarizes all of the effort and materials required to support the SCE project during its operational phase. It includes all ground operation and STS integration activities, flight and mission operations, and operations support.

p. WBS 1.8 - Program Management. This WBS element summarizes all of the effort required to manage, direct, and control the entire SCE program. These functional tasks and activities include planning, organizing, budgeting, scheduling, directing, and controlling other administrative tasks to ensure the overall objectives of the program are accomplished.

6.1.4 FLIGHT EXPERIMENT COST ESTIMATES. Following the selection of the preferred concept from the candidates examined in the first phase of the study, additional analysis provided increased design definition detail and refined input parameters used in the cost analysis. Using the updated information concerning the current SCE configuration generated in this phase of study, new cost estimates were made for the selected SCE as defined. The results of this analysis are presented in Table 6-1. The total cost for the design, development, fabrication, and test of the SCE is approximately $12M. The experiment flight hardware fabrication accounts for about $5.3M and the remaining $6.9M is required for design and analysis, component development and test, system engineering, the system level test, program, and program management. It should be noted that all system level testing and integration is conducted using the flight experiment equipment that is subsequently refurbished for flight configuration. Also included in this design and development cost is software at $0.2M, GSE at $0.16M, and spare and repair parts at $0.27M.

Table 6-1. Preliminary ROM Cost Estimates

Item	COST (1982 M$)	
	Design & Development	Fabrication
Flight Hardware		
Structure	1.96	2.86
Dynamic Test Equipment	0.93	1.01
RMS/EVA Test Equipment	0.14	0.20
Flight Support Equipment	1.30	0.60
Assembly, Integration, and C/O	—	0.27
Software	0.20	—
System Engineering & Integration	0.77	—
System Test	0.78	0.13
GSE	0.16	—
Spares	0.27	—
Facilities	0	
Program Management	0.34	0.25
TOTAL	6.85	5.32
GRAND TOTAL	12.17	

The majority of the hardware design and development cost is required for structure and mechanisms including the truss, its deployment mechanism, and the supporting structure (FSE) for mounting the SCE in the Shuttle payload bay. The dynamic test equipment is considered as virtually all off-the-shelf equipment such as gyros and accelerometers and very little in the way of component development will be required. Only a nominal cost allowance is required for the RMS/EVA test equipmetn in that there are mass and form mockups only to establish the feasibility of attaching equipment to the truss beam.

Operations costs were not estimated at this time but would consist of transportation (to KSC), and ground operations for preparation for STS installation and postflight disposition plus support activities during the flight.

6.1.5 ANNUAL FUNDING REQUIREMENTS. Annual funding requirements by fiscal year for development and flight article fabrication were generated by spreading individual cost elements in accordance with the subsequent program schedules discussed below. These annual funding requirements for the SCE are presented in Figure 6-2 and highlight the funding differences between the two schedule options.
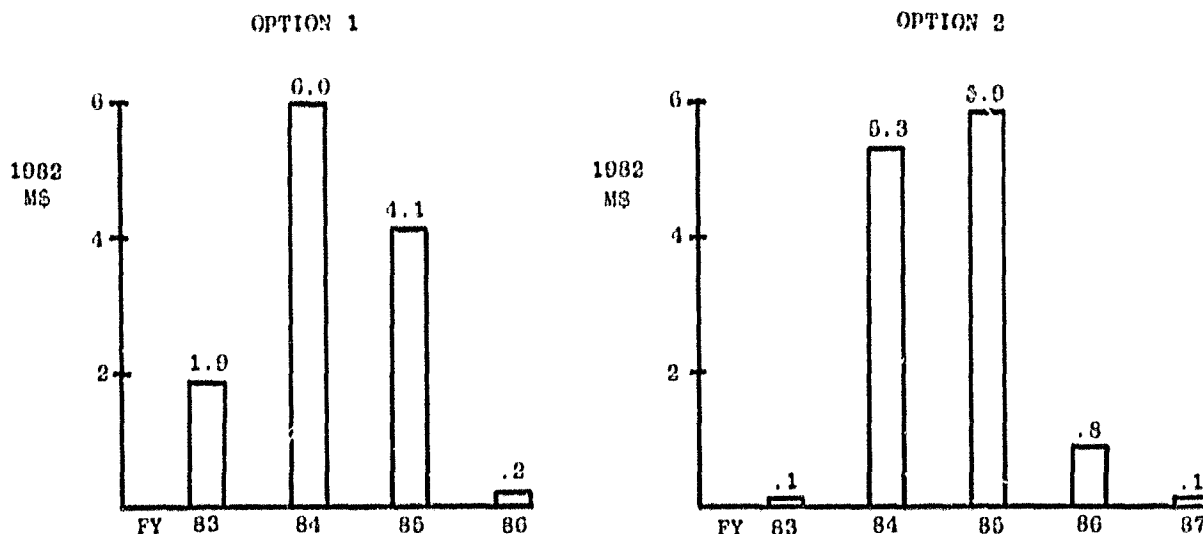
Figure 6-2. SCE Annual Funding Requirements

6.2 PROGRAM DEVELOPMENT PLAN

Based on the overall program scope of this SCE and the desired milestones, two summary program development schedules have been established. The first schedule (Figure 6-3) represents a nominal development approach keyed to a flight in late CY86. The second schedule (Figure 6-4) is designed for a slower startup and a flight one year later in CY87.

The approach used to develop these master schedules was to first establish the overall program milestones. All major functional task areas were then identified, together with the necessary sequence of major activities and events. These were to include the sequence of functions and tasks required for each of the principal phases: experiment development and test, flight article fabrication, and the operational flight. Once these major milestones and tasks were identified, detailed program milestones, task durations, and other pertinent data were laid out in the master program schedule. They key activities of each functional task area discipline are identified and time-phased relationships to each other and to the external program milestones were identified. This program master schedule serves as a focal point for displaying and evaluating interface constraints and time-critical elements.

In Option 1 the overall design and development schedule for this experiment provides for a 42-month development program leading to the flight test in November 1986. The development period is preceded by a Phase A/B definition phase in 1981 and 1982.

The planned SCE development is initiated in mid-CY83. Initial design and analysis and development milestones include a Preliminary Requirement Review (PRR) at eight weeks and a Preliminary Design Review (PDR) at six months. The Critical Design Review (CDR) follows PDR by five months. The first

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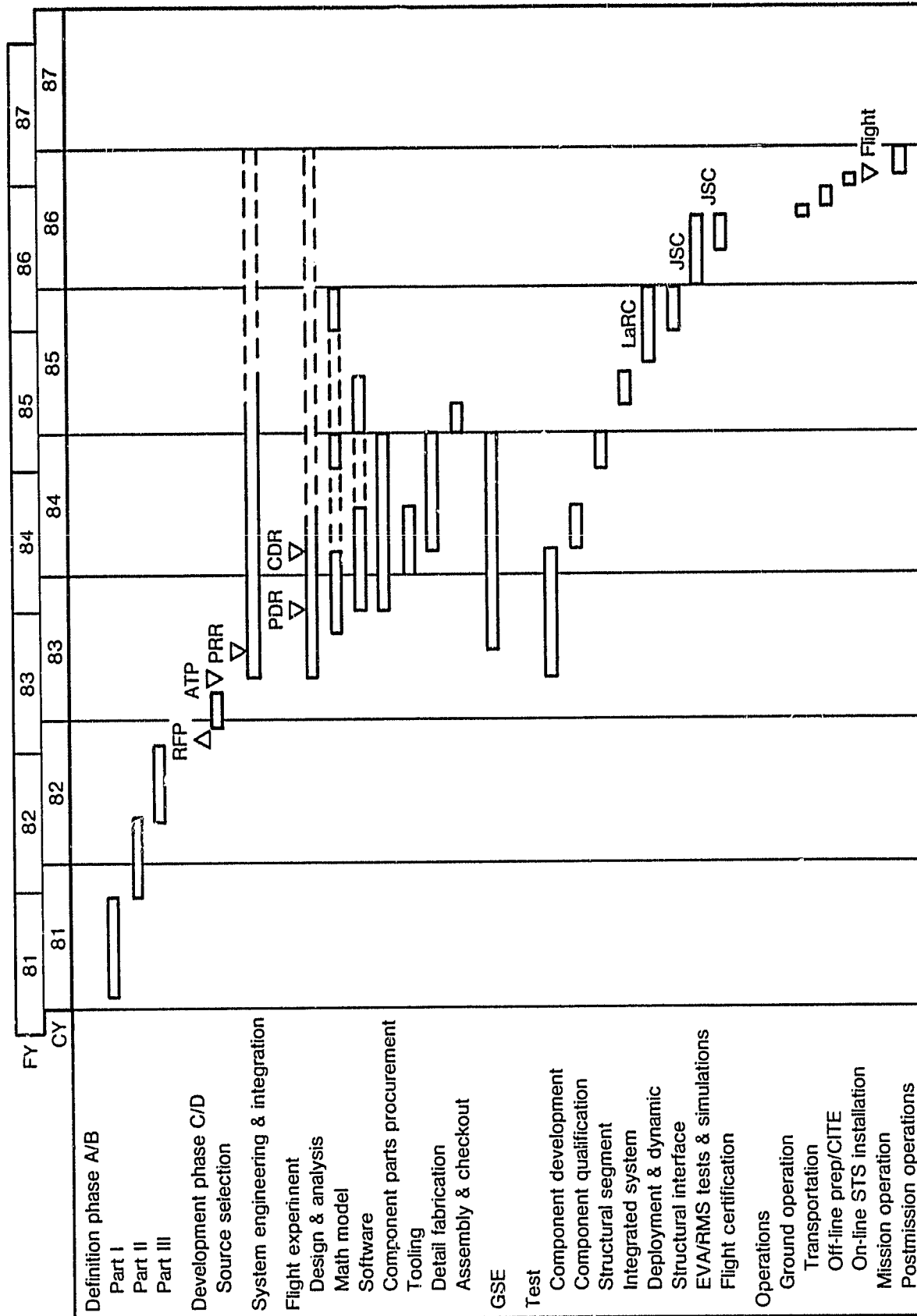
Figure 6-3. Preliminary SCE Program Development Schedule – Option 1
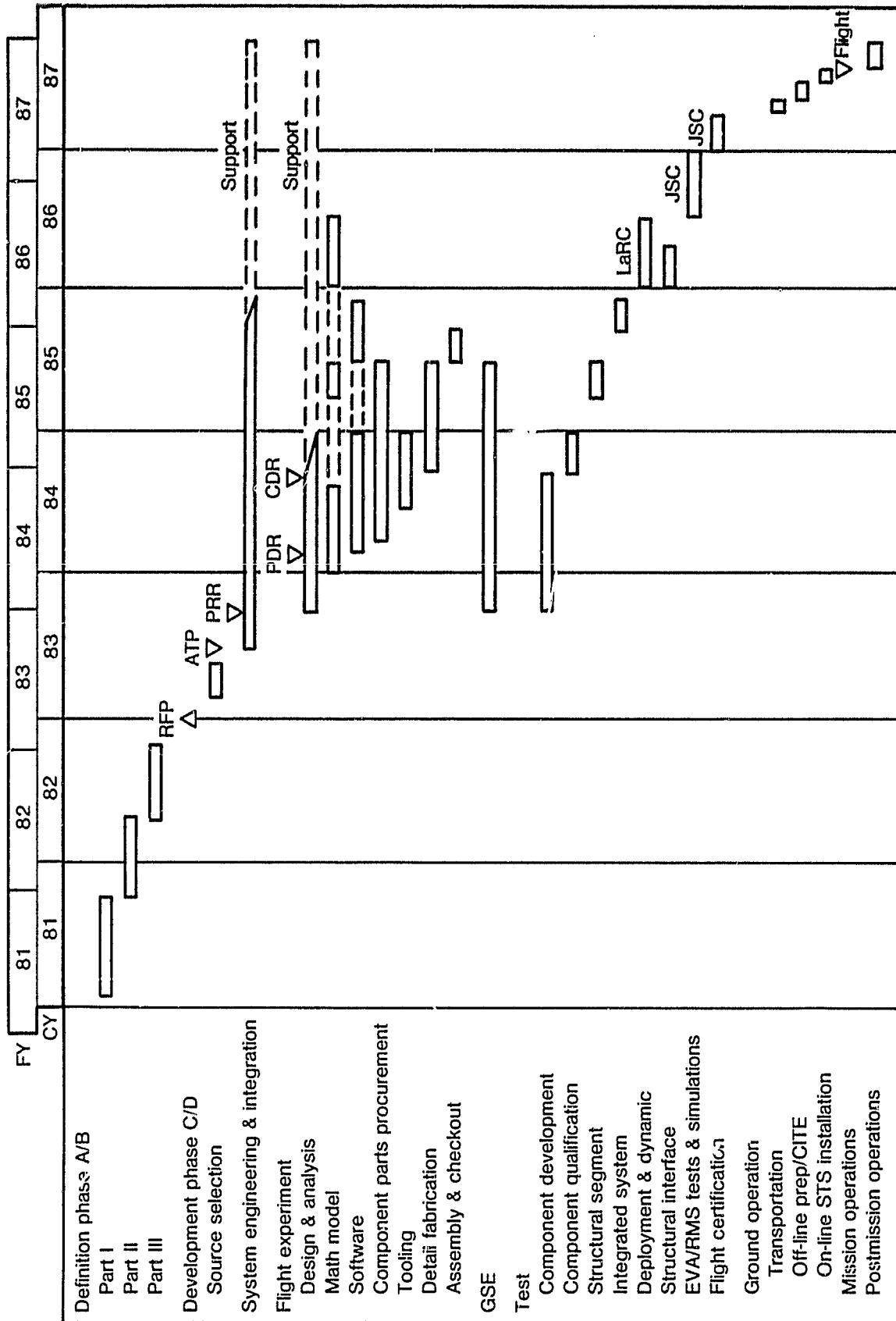
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Figure 6-4. Preliminary SCE Program Development Schedule – Option 2

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tooling is available for the parts fabrication in twelve months, and the overall experiment fabrication is completed at 23 months. Contractor development and system testing using the flight experiment hardware is completed in about 25 months, about mid-CY85. System testing of the SCE is preceded by the normal component and assembly testing in support of the development effort as well as the required qualification certifications. The SCE is then delivered to LaRC and JSC for additional system level testing.

Following NASA testing, the SCE is transported to John F. Kennedy Space Center (KSC) for a two-month period for integration processing and installation into the Space Transportation System (STS). This period may be shorter and some of the preparation may be conducted at JSC because of NASA desire to minimize STS cargo on-site residency time at KSC. This period is followed by the operational launch, deployment, and test. After return to earth, a nominal postflight time allowance is scheduled to handle disposition of flight experiment and GSE, and to analyze and evaluate the flight test data.

In Option 2, the development period has been extended to 48 months and delayed to lessen the annual funding requirements and minimize the FY83 requirements but still provide for a flight in CY87. In this option, the program go-ahead is delayed to the last quarter of FY83 and the bulk of the contractor design and development testing, and fabrication and assembly is conducted in FY84 and FY85, respectively. Major testing is accomplished in FY86 and FY87, and the flight is scheduled in the last quarter of FY87.

SECTION 7

CONCLUSIONS AND RECOMMENDATIONS

This section presents the major conclusions from the SCEDS Part II study effort and provides recommendations for subsequent program efforts.

7.1 CONCLUSIONS

7.1.1 PRELIMINARY DESIGN AND ANALYSIS

a. The basic requirements for a representative large space antenna feed mast can be satisfied with the tetrahedral deployable diamond truss as reconfigured in Part II of the study.

b. The modal excitation approach of using torque wheels at the tip of the test structure offers an excellent solution for exciting the lower modes.

c. Structural dynamic modeling accuracies are enhanced through component, subassembly, and partially deployed ground testing.

d. A flexible base mount for the test structure allows the modal characteristics to be varied so that Orbiter DAP control capabilities can be challenged by approaching its control limits by degrees.

e. Mission assignment is required to confirm the basic experiment envelope and Orbiter interfaces.

7.1.2 FLIGHT CONTROL ANALYSIS. Reduced modal frequencies of the test structure have been shown to provide a control challenge to the DAP.

7.1.3 PROGRAMMATICS

a. A 1986 flight is achievable is program start is initiated in early 1983.

b. Total SCE program costs have escalated to over $10M as a result of the changes in requirements and greater detail of definition accomplished in Part II.

7.2 RECOMMENDATIONS

7.2.1 SYSTEMS DESIGN AND ANALYSIS

a. Process request for preliminary mission assignment based on Part II results.

b. Evaluate Part II preliminary design for cost reduction approaches.

c. Further refine SCE preliminary design to incorporate cost reduction changes and mission assignment constraints.

d. Perform preliminary design of EVA/RMS experiments.

e. Perform preliminary design of potential add-on experiments such as plume effects measurements.

7.2.2 FLIGHT CONTROL ANALYSIS

a. Review CSDL DAP-structure interactions analysis data and refine modal excitation and DAP interactions amplitudes and loads.

b. Perform dynamic analysis of partially deployed case.

SECTION 8

REFERENCES

1. Space Construction Experiment Definition Study (SCEDS), Part I, Final Report, Volume II, General Dynamics Convair Division Report No. GDC-ASP-81-010, Contract No. NAS9-16303, September 1981.

2. Implementation Procedure for STS Payloads System Safety Requirements, JSC 13830, May 1979.

3. Safety Policy and Requirements for Payloads Using STS, NHB 1700.7A, December 1980.

4. Space Shuttle System Payload Accommodations, JSC 07700, Volume XIV, Rev. G, 26 September 1980.

APPENDIX A

SPACE CONSTRUCTION EXPERIMENT
PRELIMINARY SAFETY ANALYSIS

PAYLOAD SAFETY MATRIX

PAYLOAD	PAYLOAD ORGANIZATION	DATE	PAGE
SCE	GD – Convair	2-2-82	1

HAZARD GROUP / SUBSYSTEM	COLLISION	CONTAMINATION (TOXICITY, ETC)	CORROSION	ELECTRICAL SHOCK	EXPLOSION	FIRE	INJURY AND ILLNESS	LOSS OF ENTRY CAPABILITY	RADIATION	TEMPERATURE EXTREMES							
Biomedical																	
Hazard Detection and Safing*																	
Cryogenics																	
Electrical				X	X	X			X								
Environmental Control																	
Human Factors																	
Hydraulics																	
Materials		X				X											
Mechanical	X						X	X									
Optical																	
Pressure Systems																	
Propulsion																	
Pyrotechnics					X												
Radiation																	
Structures	X																
Ground Support Equipment	X																

HAZARD LIST

PAYLOAD SCE	SUBSYSTEM Electrical	DATE 2-3-82

HAZARD GROUP	HAZARD TITLE	APPLICABLE SAFETY REQUIREMENT
Electrical Shock	Personnel contact with electricity	201.2, 202.2, 215, 217
Explosion	Rupturing of electronic packages	201.2, 202.2, 208.7
Fire	Ignition of electronic packages and/or surrounding materials	201.2, 202.2, 206, 213
Temperature Extremes	Hot surface induced by excessive current flow	201.2, 202.2, 206, 213

NOTE: Unless indicated otherwise, the paragraph numbers for the requirements are taken from NHB 1700.7A "Safety Policy and Requirements for Payloads using the Space Transportation System (STS)."

A-3

HAZARD LIST

PAYLOAD SCE	SUBSYSTEM Materials	DATE 2-9-82
HAZARD GROUP	HAZARD TITLE	APPLICABLE SAFETY REQUIREMENT
Contamination	Offgassing of hazardous materials	201.2, 209.4
Fire	Flammable materials support combustion	201.2, 202.2, 206, 209.3

A-4

HAZARD LIST

PAYLOAD	SUBSYSTEM	DATE
SCE	Mechanical	2-2-82

HAZARD GROUP	HAZARD TITLE	APPLICABLE SAFETY REQUIREMENT
Collision	Premature beam extension	201.2, 202.2c2, 206, 207
	Premature jettison	201.2, 202.2c2, 205, 206, 207
	Premature release of rail latches	201.2, 202.2c2, 206, 207
	Premature release of tip mass	201.2, 202.2c2, 206, 207
	Orbiter cargo bay doors close prematurely	201.2, 202.2
	RMS collides with STS	201.2, 202.2
Injury and Illness	Inadvertent retraction during EVA	201.2, 202.2, 217
	RMS injures personnel during EVA	201.2, 202.2, 217
	Damage to space suit during EVA	201.2, 202.2, 217
Loss of Entry Capability	Payload blocks closure of cargo bay doors	201.2, 202.2d, 205
Explosion	Inadvertent initiation of pyro-technic separation bolt	210

A-5

HAZARD LIST

PAYLOAD SCE	SUBSYSTEM Structures	DATE 2-3.82
HAZARD GROUP	**HAZARD TITLE**	**APPLICABLE SAFETY REQUIREMENT**
Collision	Failure of beam structure	201.2, 2v2.2, 208.1, 208.2, 208.3
	Failure of support structure	201.2, 202.2, 208.1, 208.2, 208.3

A-6

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HAZARD LIST

PAYLOAD	SUBSYSTEM	DATE
SCE	Ground Support Equipment	1 - 82

HAZARD GROUP	HAZARD TITLE	APPLICABLE SAFETY REQUIREMENT
Collision	Loss of control during ground handling	201.2, 202.2, 213, 215
	Structural failure of GSE	201.2, 202.2, 208.1., 208.3

A-7

PAYLOAD HAZARD REPORT		NO. E-1

PAYLOAD SCE	PHASE 0
SUBSYSTEM Electrical	DATE 2-3-82

HAZARD TITLE
Electrical Shock   Group: Electrical Shock

APPLICABLE SAFETY REQUIREMENTS:

201.2 - Failure Tolerance, Catastrophic Hazards   215 - Hazardous Procedures
202.2 - Control of Hazardous Functions, Catastrophic 217 - Extravehicular Activity

DESCRIPTION OF HAZARD:

Personnel contacts exposed, live, electrical conductor and ground.

HAZARD CAUSES:

Wire harness damage exposes conductors.

HAZARD CONTROLS:

1. Only 28 VDC available, no health hazard.

2. Power not applied on ground except during test.

SAFETY VERIFICATION METHODS:

STATUS:

Open - pending detailed design and safety review

CONCURRENCE	PHASE I	PHASE II
Payload Organization		
STS Operator		

APPROVAL	PHASE III	
Payload Organization		STS Operator

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PAYLOAD HAZARD REPORT

	NO.
	E-2

PAYLOAD	PHASE
SCE	0

SUBSYSTEM	DATE
Electrical	2-82

HAZARD TITLE

Rupturing of Electronic Packages   Group: Explosion

APPLICABLE SAFETY REQUIREMENTS:

201.2 - Failure Tolerance, Castastrophic Hazards
202.2 - Control of Hazardous Funcrions, Catastrophic
208.7 - Structural, Sealed Containers

DESCRIPTION OF HAZARD:

Explosion of electronic packages can damage other equipment or injury personnel. Damage could affect control of safety functions.

HAZARD CAUSES:

Pressure buildup in packages if no venting system is provided.

HAZARD CONTROLS:

Electronic packages will be fully ventilated or sealed. Sealed packages will be proof tested to demonstrate pressure integrity.

SAFETY VERIFICATION METHODS:

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STATUS:

Open - pending detailed design and safety review

CONCURRENCE	PHASE I	PHASE II
Payload Organization		
STS Operator		

APPROVAL	PHASE III	
Payload Organization	STS Operator	

PAYLOAD HAZARD REPORT		NO. E-3

PAYLOAD SCE	PHASE 0
SUBSYSTEM Electrical	DATE 2-82

HAZARD TITLE Ignition of Electronic Packages   Group: Fire

APPLICABLE SAFETY REQUIREMENTS:
201.2 - Failure Tolerance, Catastrophic Hazards   206 - Failure Propagation
202.2 - Control of Hazardous Functions, Catastrophic   213 - Electrical Systems

DESCRIPTION OF HAZARD:

Overheating of electrical component causes ignition of insulation and/or surrounding materials. Electronic components ignite flammable vapors.

HAZARD CAUSES:

1. Excessive current in harness causes overheating.

2. Arci..g, sparking, or hot spots cause ignition of vapors.

HAZARD CONTROLS:

1. Provide current limiting to limit fault currents to safe values.

2. Perform explosuve atmosphere qualification tests of electronic assemblies.

SAFETY VERIFICATION METHODS:

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STATUS:

Open - pending detailed design and safety review

CONCURRENCE	PHASE I	PHASE II
Payload Organization		
STS Operator		

APPROVAL	PHASE III	
Payload Organization		STS Operator

PAYLOAD HAZARD REPORT		NO. E-4

PAYLOAD SCE	PHASE 0

SUBSYSTEM Electrical	DATE

HAZARD TITLE
Hot Surface Induced by Excessive Current Flow   Group: Temperature Extremes

APPLICABLE SAFETY REQUIREMENTS:

201.2 – Failure Tolerance, Catastrophic Hazards   206 – Failure Propagation
202.2 – Control of Hazardous Functions, Catastrophic   213 – Electrical Systems

DESCRIPTION OF HAZARD:

Excessive current flow through wires, structural elements, or tools creates a hot surface.

HAZARD CAUSES:

1. Short circuit in electronic package or E/M device.

2. Short circuit through structural member or tool.

HAZARD CONTROLS:

1. Provide current limiting to limit fault currents to safe values.

2. Power not applied on ground except during test.

3. Power shall be inhibited during EVA contact with experiment.

SAFETY VERIFICATION METHODS:

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STATUS:

Open – pending detailed design and safety review

CONCURRENCE	PHASE I	PHASE II
Payload Organization		
STS Operator		

APPROVAL	PHASE III	
Payload Organization	STS Operator	

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PAYLOAD HAZARD REPORT

PAYLOAD SCE	**NO.** M-1
	PHASE 0
SUBSYSTEM Materials	**DATE** 2-82

HAZARD TITLE
Offgassing of Hazardous Materials   Group: Contamination

APPLICABLE SAFETY REQUIREMENTS:

201.2 - Failure Tolerance, Catastrophic Hazards
209.4 - Material Offgassing

DESCRIPTION OF HAZARD:

Objectionable odors or foul offgassing from materials used in equipment on aft flight deck.

HAZARD CAUSES:

Improper material selection and control.

HAZARD CONTROLS:

There is no unique aft flight deck equipment necessary for the experiment operations. The standard Orbiter supplied switch panel will be used. This equipment will be flight-proven prior to this mission.

SAFETY VERIFICATION METHODS:

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STATUS:

Open. Recommend hazard report closeout based on use of established Orbiter equipment.

CONCURRENCE	PHASE I	PHASE II
Payload Organization		
STS Operator		

APPROVAL	PHASE III	
Payload Organization		STS Operator

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PAYLOAD HAZARD REPORT

	NO. M-2

PAYLOAD SCE	PHASE 0

SUBSYSTEM Materials	DATE 2-82

HAZARD TITLE
Flammable Materials Support Combustion   Group: Fire

APPLICABLE SAFETY REQUIREMENTS:
201.2 - Failure Tolerance, Catastrophic Hazards   206 - Failure Propagation
202.2 - Control of Hazardous Functions, Catastrophic   209.3 - Flammable Materials

DESCRIPTION OF HAZARD:

Combustion in the Orbiter cargo bay is sustained by SCE materials.

HAZARD CAUSES:

Exposure of flammable materials to ignition sources.

HAZARD CONTROLS:

1. Eliminate/minimize ignition sources.

2. Use materials shown to be non-flammable or self-extinguishing where practicable.

SAFETY VERIFICATION METHODS:

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STATUS:

Open - pending detailed design and safety review

CONCURRENCE	PHASE I	PHASE II
Payload Organization		
STS Operator		

APPROVAL	PHASE III	
Payload Organization	STS Operator	

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PAYLOAD HAZARD REPORT		NO. Me-1
PAYLOAD SCE		PHASE 0
SUBSYSTEM Mechanical		DATE 2-3-82

HAZARD TITLE
Premature Beam Extension   Group: Collision

APPLICABLE SAFETY REQUIREMENTS:
201.2 – Failure Tolerance, Catastrophic Hazards   206 – Failure Propagation
202.2c2 – Control of Deployment Function   207 – Redundancy Separation

DESCRIPTION OF HAZARD:

Inadvertent extension of truss assembly prior to truss package rotation results in collision with Orbiter.

HAZARD CAUSES:

1.   Inadvertent command to extension mechanism.

2.   Failure of truss retaining latches.

HAZARD CONTROLS:

Deployment control must be 2 F/T to prevent inadvertent truss extension.

SAFETY VERIFICATION METHODS:

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STATUS:

Open – pending detailed design and safety review

CONCURRENCE	PHASE I	PHASE II
Payload Organization		
STS Operator		

APPROVAL	PHASE III	
Payload Organization	STS Operator	

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PAYLOAD HAZARD REPORT

NO.	Me-2

PAYLOAD	PHASE
SCE	O

SUBSYSTEM	DATE
Mechanical	2-3-82

HAZARD TITLE
Premature Release of Jettison Latches   Group: Collision

APPLICABLE SAFETY REQUIREMENTS:

201.1 – Failure Tolerance, Catastrophic Hazards   206 – Failure Propagation
202.2c2 – Control of Deployment Function   207 – Redundancy Separation

DESCRIPTION OF HAZARD:

Inadvertent release of jettison latches prior to tip mass ejection
and RMS attachment results in collision between SCE and Orbiter.

HAZARD CAUSES:

Inadvertent release command to the jettison latches.

HAZARD CONTROLS:

Jettison latches are Shuttle-provided and will be flight-proven
prior to this mission.

SAFETY VERIFICATION METHODS:

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STATUS:

Open.   Recommend hazard report closeout based on use of established
Orbiter equipment.

CONCURRENCE	PHASE I	PHASE II
Payload Organization		
STS Operator		

APPROVAL	PHASE III	
Payload Organization		STS Operator

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PAYLOAD HAZARD REPORT

	NO. Me-3
PAYLOAD SCE	**PHASE** 0
SUBSYSTEM Mechanical	**DATE** 2-3-82
HAZARD TITLE Premature Release of Rail Latches	Group: Collision

APPLICABLE SAFETY REQUIREMENTS:

201.2 - Failure Tolerance, Catastrophic Hazards   206 - Failure Propagation
202.2c2 - Control of Deployment Function   207 - Redundancy Separation

DESCRIPTION OF HAZARD:

Inadvertent release of rail latches prior to cargo bay door opening results in collision with Orbiter.

HAZARD CAUSES:

1. Structural failure of rail latch mechanisms.

2. Inadvertent command to open rail latches.

HAZARD CONTROLS:

1. Appropriate structural safety factors should be applied to design of rail latch mechanism.

2. Mechanism control must be 2 F/T to prevent inadvertent rail latch release.

SAFETY VERIFICATION METHODS:

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STATUS:

Open - pending detailed design and safety review

CONCURRENCE	PHASE I	PHASE II
Payload Organization		
STS Operator		

APPROVAL	PHASE III	
Payload Organization		STS Operator

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PAYLOAD HAZARD REPORT

		NO. Mc-4

PAYLOAD SCE		PHASE 0
SUBSYSTEM Mechanical		DATE 2-3-82

HAZARD TITLE
Premature Release of Tip Mass   Group: Collision

APPLICABLE SAFETY REQUIREMENTS:

201.2 - Failure Tolerance, Catastrophic Hazards   206 - Failure Propagation
202.2c2 - Control of Deployment Function   207 - Redundancy Separation

DESCRIPTION OF HAZARD:

Inadvertent release of tip mass prior to truss package rotation
results in collision with Orbiter.

HAZARD CAUSES:

1. Structural failure of tip mass anchoring mechanism.

2. Inadvertent command to release tip masses.

HAZARD CONTROLS:

1. Appropriate structural safety factors should be applied to
   design of tip mass anchoring mechanism.

2. Mechanism control must be 2 F/T to prevent inadvertent
   release of tip masses.

SAFETY VERIFICATION METHODS:

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STATUS:

Open - pending detailed design and safety review

CONCURRENCE	PHASE I	PHASE II
Payload Organization		
STS Operator		

APPROVAL	PHASE III	
Payload Organization		STS Operator

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PAYLOAD HAZARD REPORT		NO. Me-5
PAYLOAD SCE		PHASE 0
SUBSYSTEM Mechanical		DATE

HAZARD TITLE
Orbiter Cargo Bay Doors Close Prematurely   Group: Collision

APPLICABLE SAFETY REQUIREMENTS:

201.2 - Failure Tolerance, Catastrophic Hazards
202.2 - Control of Hazardous Functions, Catastrophic

DESCRIPTION OF HAZARD:

Orbiter cargo bay doors inadvertently close while truss is deployed.

HAZARD CAUSES:

Inadvertent command to close bay doors.

HAZARD CONTROLS:

Cargo bay doors are Shuttle-provided and will be flight-proven prior
to this mission.

SAFETY VERIFICATION METHODS:

STATUS:

Open. Recommend hazard report closeout based on use of established
Orbiter equipment.

CONCURRENCE	PHASE I	PHASE II
Payload Organization		
STS Operator		

APPROVAL	PHASE III	
Payload Organization	STS Operator	

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PAYLOAD HAZARD REPORT

	NO.
	Mc-6

PAYLOAD	PHASE
SCE	0

SUBSYSTEM	DATE
Mechanical	

HAZARD TITLE
RMS Collides with Orbiter   Group: Collision

APPLICABLE SAFETY REQUIREMENTS:

201.2 - Failure Tolerance, Catastrophic Hazards
202.2 - Control of Hazardous Functions, Catastrophic

DESCRIPTION OF HAZARD:

RMS collides with Orbiter during experiment operations.

HAZARD CAUSES:

1. Loss of control of RMS

2. Improper command to RMS

3. Structural failure of RMS

4. Inadvertent RMS operations while doors are closed

HAZARD CONTROLS:

RMS is Shuttle-provided and will be flight-proven prior to this mission.

SAFETY VERIFICATION METHODS:

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STATUS:

Open. Recommend hazard report closeout based on use of established
Orbiter equipment.

CONCURRENCE	PHASE I	PHASE II
Payload Organization		
STS Operator		

APPROVAL	PHASE III	
Payload Organization		STS Operator

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PAYLOAD HAZARD REPORT		NO. Mc-7

PAYLOAD SCE	PHASE 0

SUBSYSTEM Mechanical	DATE 2-3-82

HAZARD TITLE
Inadvertent Truss Retraction During EVA   Group: Injury and Illness

APPLICABLE SAFETY REQUIREMENTS:
201.2 - Failure Tolerance, Catastrophic Hazards
202.2 - Control of Hazardous Functions, Catastrophic
217   - Extravehicular Activity

DESCRIPTION OF HAZARD:

Inadvertent retraction of bay while personnel is near causes injury.

HAZARD CAUSES:

Inadvertent retract command to deployment mechanism during EVA.

HAZARD CONTROLS:

Deployment control must be 2 F/T to prevent inadvertent retraction.

SAFETY VERIFICATION METHODS:

OF
OF

STATUS:

Open - pending detailed design and safety review

CONCURRENCE	PHASE I	PHASE II
Payload Organization		
STS Operator		

APPROVAL	PHASE III	
Payload Organization		STS Operator

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PAYLOAD HAZARD REPORT

	NO. Mc-8

PAYLOAD	PHASE
SCE	0

SUBSYSTEM	DATE
Mechanical	2-82

HAZARD TITLE RMS Injures Personnel During EVA   Group: Injury and Illness

APPLICABLE SAFETY REQUIREMENTS:

201.2 - Failure Tolerance, Catastrophic Hazards
202.2 - Control of Hazardous Functions, Catastrophic
217   - Extravehicular Activity

DESCRIPTION OF HAZARD:

RMS strikes or pushes personnel during EVA causing injury.

HAZARD CAUSES:

1. Loss of control of RMS

2. Improper command to RMS

3. Structural failure of RMS

HAZARD CONTROLS:

RMS is Shuttle-provided and will be flight-proven prior to this mission.

SAFETY VERIFICATION METHODS:

STATUS:

Open. Recommend hazard report closeout based on use of established Orbiter equipment.

CONCURRENCE	PHASE I	PHASE II
Payload Organization		
STS Operator		

APPROVAL	PHASE III	
Payload Organization		STS Operator

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PAYLOAD HAZARD REPORT		NO. Mc-9
PAYLOAD SCE		PHASE 0
SUBSYSTEM Mechanical		DATE 2-82

HAZARD TITLE Damage to Space Suit During EVA	Group: Injury and Illness

APPLICABLE SAFETY REQUIREMENTS:
201.2 - Failure Tolerance, Catastrophic Hazards
202.2 - Control of Hazardous Functions, Catastrophic
217   - Extravehicular Activity

DESCRIPTION OF HAZARD:

Damage occurring to space suit during EVA causes loss of life support and subsequent injury to personnel.

HAZARD CAUSES:

1. Sharp edges

2. Protrusions

3. Moving parts

4. Insufficient illumination

HAZARD CONTROLS:

1. All surfaces and edges must be smooth, rounded, and free of burrs as defined in Table 4-4 of JSC-10615, "Shuttle EVA Description and Design Criteria."

2. Inadvertent motion of mechanical devices should be inhibited in a 2 F/T manner.

(Continued on attached page)

SAFETY VERIFICATION METHODS:

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STATUS:

Open - pending detailed design and safety review

CONCURRENCE	PHASE I	PHASE II
Payload Organization		
STS Operator		

APPROVAL	PHASE III	
Payload Organization	STS Operator	

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Payload - SCE   Me-9
Subsystem - Mechanical
Hazard Title - Damage to Space Suit During EVA   Group: Injury and Illness

HAZARD CONTROLS: (Continued)

3. Sufficient illumination must be provided for EVA work and
    translation during the "dark" side of the earth orbit.

PAYLOAD HAZARD REPORT

NO.	Me-10

PAYLOAD SCE	**PHASE** 0
SUBSYSTEM Mechanical	**DATE** 2-3-82

HAZARD TITLE
Payload Blocks Cargo Bay Doors   Group: Loss of Entry Capability

APPLICABLE SAFETY REQUIREMENTS:
201.2 – Failure Tolerance, Catastrophic Hazards
202.2d – Deployment Preventing Door Closure
205   – Contingency Return of Payloads

DESCRIPTION OF HAZARD:

Orbiter unable to close cargo bay doors for re-entry because
SCE payload blocks opening.

HAZARD CAUSES:

1.   Unable to retract truss and unable to jettison SCE payload package.

2.   Possible causes of loss of jettison capability are: Orbiter latch
     failure, RMS failure, or failure to release tip masses.

HAZARD CONTROLS:

The combination of retraction and jettison must be 2 failure tolerant
to comply with requirements.

SAFETY VERIFICATION METHODS:

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OF POOR QUALITY

STATUS:

Open – pending detailed design and safety review

CONCURRENCE	PHASE I	PHASE II
Payload Organization		
STS Operator		

APPROVAL	PHASE III	
Payload Organization		STS Operator

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PAYLOAD HAZARD REPORT

NO. Me-11

PAYLOAD SCE

PHASE 0

SUBSYSTEM Mechanical

DATE 4-29-82

HAZARD TITLE Inadvertent Initiation of Pyrotechnic Separation Bolt

APPLICABLE SAFETY REQUIREMENTS:
210 - Pyrotechnics

DESCRIPTION OF HAZARD:
Inadvertent detonation of an electro explosive device could result in an explosion because of direct or secondary effects.

    a. Early initiation after installation would result in release of the tip mass which is discussed in Report Me-4

    b. Failure of the separation bolt to operate would result in payload bay door blockage discussed in Report Me-10

HAZARD CAUSES:

    1. Inadvertent initiation of electro explosive device from RFI/electrostatic discharge

    2. Premature initiation of electro explosive device due to premature power application.

HAZARD CONTROLS:

    1. The electro explosive device used for the system must be the NASA standard initiator (NSI)

    2. The control unit harness must be tested for stray voltage prior to NSI connection

    3. The command circuit must be 2 F/T to prevent inadvertent initiation

SAFETY VERIFICATION METHODS:

STATUS:

    Open - pending detailed design and safety review

CONCURRENCE	PHASE I	PHASE II
Payload Organization		
STS Operator		

APPROVAL	PHASE III	
Payload Organization		STS Operator

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PAYLOAD HAZARD REPORT	NO. S-1

PAYLOAD SCE	PHASE 0

SUBSYSTEM Structures	DATE 2-82

HAZARD TITLE
Failure of Beam Structure   Group: Collision

APPLICABLE SAFETY REQUIREMENTS:

201.2 – Failure Tolerance Catastrophic Hazards   208.2 – Emergency Landing Loads
202.2 – Control of Hazardous Functions, Catastrophic   208.3 – Stress Corrosion
208.1 – Structural Design

DESCRIPTION OF HAZARD:

Structural failure of beam causes collision with Orbiter.

HAZARD CAUSES:

1. Material defects

2. Stress corrosion

3. Corrosion due to dissimilar metals and exposure

4. Undetected damage

5. Load distribution not conforming to analytic prediction causing local overstress

6. Failure of hinge to lock

7. Impact by RMS

HAZARD CONTROLS:

1. A fracture control analysis should be performed to preclude failures caused by propagation of existing flaws.

2. Whenever practical stress corrosion resistant materials (per MFSC – Spec – 522A) should be used.

(Continued on attached page)

SAFETY VERIFICATION METHODS:

STATUS:

Open – pending detailed design and safety review

CONCURRENCE	PHASE I	PHASE II
Payload Organization		
STS Operator		

APPROVAL	PHASE III	
Payload Organization		STS Operator

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PAYLOAD HAZARD REPORT

Payload - SCE
Subsystem - Structures
Hazard Title - Failure of Beam Structure   Group: Collision

HAZARD CONTROLS: (Continued)

3. Impression marking of parts should be prohibited to minimize
   potential stress corrosion.

4. Faying surfaces of dissimilar metal should receive surface
   protection to eliminate corrosion.

5. Appropriate structural safety factors should be applied to
   the design of the beam.

6. Beam geometry is 1 F/T.

7. RMS is Shuttle-provided and will be flight-proven prior to
   this mission.

PAYLOAD HAZARD REPORT

	NO. S-2

PAYLOAD SCE	PHASE 0

SUBSYSTEM Structures	DATE 2-82

HAZARD TITLE
Failure of Support Structure   Group: Collision

APPLICABLE SAFETY REQUIREMENTS:
201.2 - Failure Tolerance, Catastrophic Hazards   208.2 - Emergency Landing Loads
202.2 - Control of Hazardous Functions, Catastrophic   208.3 - Stress Corrosion
208.1 - Structural Design

DESCRIPTION OF HAZARD:

Structural failure of beam support structure causes collision
with Orbiter.

HAZARD CAUSES:

1. Material defects

2. Stress corrosion

3. Corrosion due to dissimilar
    metals and exposure

4. Undetected damage

5. Load distribution not conforming
    to analytic prediction causing
    local overstress

6. Impact by RMS

HAZARD CONTROLS:

1. A fracture control analysis should be performed to preclude failures
    caused by propagation of existing flaws.

2. Whenever practical, stress corrosion resistant materials (per MFSC -
    Spec - 522A) should be used.

                    (Continued on attached page)

SAFETY VERIFICATION METHODS:

STATUS:

Open - pending detailed design and safety review

CONCURRENCE	PHASE I	PHASE II
Payload Organization		
STS Operator		

APPROVAL	PHASE III	
Payload Organization		STS Operator

Payload - SCE
S-2
Subsystem - Structures
Hazard Title - Failure of Support Structure   Group: Collision

HAZARD CONTROLS: (Continued)

3. Impression marking of parts should be prohibited to minimize potential stress corrosion.

4. Faying surfaces of dissimilar metal should receive surface protection to eliminate corrosion.

5. Appropriate structural safety factors should be applied to the design of the support structure.

6. RMS is Shuttle-provided and will be flight-proven prior to this mission.

PAYLOAD HAZARD REPORT

	NO. G-1

PAYLOAD	PHASE
SCE	0

SUBSYSTEM	DATE
Ground Support Equipment	2-82

HAZARD TITLE
Ground Handling   Group: Collision

APPLICABLE SAFETY REQUIREMENTS:

201.2 – Failure Tolerance, Catastrophic Hazards   213 – Electrical Systems
202.2 – Control of Hazardous Functions   215 – Hazardous Procedures

DESCRIPTION OF HAZARD:

Loss of control during ground handling of payload causes collision with STS equipment or personnel.

HAZARD CAUSES:

Uncontrolled movement during installation into the Orbiter.

HAZARD CONTROLS:

1. Perform analysis of landing procedures to insure operations will not jeopardize the Orbiter.

2. Ground equipment is Shuttle-provided.

SAFETY VERIFICATION METHODS:

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STATUS:

Open – pending detailed design and safety review

CONCURRENCE	PHASE I	PHASE II
Payload Organization		
STS Operator		

APPROVAL	PHASE III	
Payload Organization		STS Operator

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PAYLOAD HAZARD REPORT	NO. G-2

PAYLOAD SCE	PHASE O

SUBSYSTEM Ground Support Equipment	DATE 2-82

HAZARD TITLE
Structural Failure of GSE   Group: Collision

APPLICABLE SAFETY REQUIREMENTS:
201.2 - Failure Tolerance, Catastrophic Hazards   208.1 - Structural Design
202.2 - Control of Hazardous Functions, Catastrophic   208.3 - Stress Corrosion

DESCRIPTION OF HAZARD:

Structural failure of ground support equipment causes collision with STS equipment or personnel.

HAZARD CAUSES:

1. Material defects

2. Stress corrosion

3. Corrosion due to dissimilar metals and exposure

4. Undetected damage

5. Load distribution not conforming to analytic prediction causing local overstress

HAZARD CONTROLS:

1. A fracture control analysis should be performed to preclude failures caused by propagation of existing flaws.

2. Whenever practical, stress corrosion resistant materials (per MFSC - Spec - 522A) should be used.

(Continued on attached page)

SAFETY VERIFICATION METHODS:

ORIGINAL PAGE IS
OF POOR QUALITY

STATUS:

Open - pending detailed design and safety review

CONCURRENCE	PHASE I	PHASE II
Payload Organization		
STS Operator		

APPROVAL	PHASE III	
Payload Organization	STS Operator	

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Payload - SCE
Subsystem - Ground Support Equipment
Hazard Title - Structural Failure of GSE   Group: Collision

HAZARD CONTROLS: (Continued)

3. Impression marking of parts should be prohibited to minimize potential stress corrosion.

4. Faying surfaces of dissimilar metal should receive surface protection to eliminate corrosion.

5. Appropriate structural safety factors should be applied to the design of the support structure.

6. Ground equipment is Shuttle-provided.